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The Space Engineering program at Utah State University has developed a small satellite, known as USUSat, under funding from AFOSR, AFRL, NASA and Utah State University's Space Dynamics Laboratory. This satellite was designed and significantly manufactured by students in the Mechanical and Aerospace Engineering and the Electrical and Computer Engineering Departments within the College of Engineering.

USUSat is one of three spacecraft being designed for the Ionospheric Observation Nanosatellite Formation (ION-F). This formation comprises three 15 kg. spacecraft designed and built in cooperation by Utah State University, Univcrsity of Washington, and Virginia Polytechnic Institute. The ION-F satellites are being designed and built by students at the three universities, with close coordination to insure compatibility for launch, deployment, and the formation flying mission. The JON-F mission is part of the U.S. Air Force Research Laboratory (AFRL) University Nanosatellite Program, which provides technology development and demonstrations for the TechSat2l Program. The University Nanosatellite Program involves 10 universities building nanosatellites for a launch in 2004 on two separate space shuttle missions. Additional support for the formation flying demonstration has been provided by NASA's Goddard Space Flight Center.

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Final Contract Report
To
AFRL

University Nanosatellite Program

ION-F Constellation

USUSAT

Principal Investigators:

Dr. Charles Swenson
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Dr. Frank Redd

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EXECUTIVE SUMMARY

The Space Engineering program at Utah State University has developed a small satellite, known as USUSat, under funding from AFOSR, AFRL, NASA and Utah State University's Space Dynamics Laboratory. This satellite was designed and significantly manufactured by students in the Mechanical and Aerospace Engineering and the Electrical and Computer Engineering Departments within the College of Engineering.

USUSat is one of three spacecraft being designed for the Ionospheric Observation Nanosatellite Formation (ION-F). This formation comprises three 15 kg. spacecraft designed and built in cooperation by Utah State University, University of Washington, and Virginia Polytechnic Institute. The ION-F satellites are being designed and built by students at the three universities, with close coordination to insure compatibility for launch, deployment, and the formation flying mission. The ION-F mission is part of the U.S. Air Force Research Laboratory (AFRL) University Nanosatellite Program, which provides technology development and demonstrations for the TechSat21 Program. The University Nanosatellite Program involves 10 universities building nanosatellites for a launch in 2004 on two separate space shuttle missions. Additional support for the formation flying demonstration has been provided by NASA's Goddard Space Flight Center.

Mission Objectives

The objectives of this formation are scientific research, formation flying research, technology demonstration and education. The primary scientific objective for this mission is to investigate global ionospheric effects which impact the performance of space-based radar's and other distributed satellite measurements. This requires the three spacecraft to make simultaneous, spatially distributed ionospheric plasma electron density measurements. In addition, measurements from the GPS system will be used to make the first global multi-baseline RF-scintillation measurements of the ionosphere. The scintillation of GPS signals using receivers on each spacecraft will provide information about the scale sizes of disturbances between the nanosatellite constellation and the GPS transmitter.

The ION-F formation will be used as a space-based distributed control testbed for active formation control using inter-satellite communications. Autonomous formation maneuvering and control will be performed. Maneuvers to be tested include controlling the in-track separation distance between spacecraft in the same orbit in a leader follower approach, maneuvering into common ground track orbits, and side by side operation. Positional feedback between spacecraft will be performed using a combined GPS and cross-link communications system developed by the Applied Physics Laboratory at Johns Hopkins University. Notice that the two scientific measurement experiments are enhanced by formation flying and place only limited constraints on the performance of any maneuvers.

Several new components and hardware concepts will be tested on the ION-F spacecraft. These include:

- The Applied Physics Laboratory GPS/inter-satellite "cross-link" communications system. This system will provide continuous communications on the spacecraft location within the formation as well as limited command and control between the spacecraft themselves.
- A controlled permanent magnet torquing method for attitude control. High strength rare-earth magnets are positioned using a gimbal system to generate magnetic torques on the spacecraft, requiring significantly less energy than equivalent strength torquer coils during maneuvering.
- Experiments in modulating the aerodynamic force vector for orbital control will be performed. Maneuvering the spacecraft attitude so that different cross sectional areas vary the effective ballistic coefficient for in-track maneuvers. Tests to determine whether small cross-track maneuvers can be achieved will also be made.
- Commercial CMOS cameras will be used for low power attitude measurements. Multiple cameras positioned around the spacecraft will be used for determining both horizon locations and sun position. The cameras pixel array will be directly memory mapped into the command processor.
- An internet based operations center will be developed to allow control of each satellite from the appropriate campus location. Ground site locations will be in Logan, Utah and Blacksburg, Virginia.
- A low mass separation system developed by Planetary Systems Inc. will be tested for inter-satellite separation.
- A new Air Force platform known as the Multiple Satellite Deployment System (MSDS) designed to work with the Space Shuttle Shels release system will be tested.

Educational Objectives

This program brings a unique, hand-on spacecraft design experience to undergraduate and graduate students beyond that taught in traditional spacecraft design courses. USUSat has achieved an impressive record of student participation with over 22 graduate students and 25 undergraduate students having worked on this project for at least one semester. A list of these students is provided in the following list. Further, this list shows a significant number of theses, dissertations papers and reports have been written on USUSat/IONF.

Students		Area	Status	Thesis	Report
Allan	Kimberly	Software	Undergraduate student		
Ashby	Bret	Structures	Graduate student	X	
Ballard	Doug	Wiring Harness	Engineer		
Barjatya	Aroh	Camera Electronics	Graduate student	X	
Brainard	Doug	Flight Computer	Undergraduate student		X
Chandrasekaran	Anuradha	Communications	Graduate student	X	X
Crocker	Barry	Thermal	Undergraduate student		X
Fish	Chad	Science / Power	Doctoral student	X	
Florin	Dominic	Attitude Control	Graduate student		X
Fullmer	Rees	Co-Principal Investigator	Professor		
Gettamaneni	Kumaresh	Booms	Graduate student		X
Goudie	Tyler	Power	Undergraduate student		X
Gunda	Siva	Documentation	Undergraduate student		X
Gutshall	James	Communications	Undergraduate student		X
Harmon	Michael	Antennas	Undergraduate student		X
Harris	James	Thermal Systems	Undergraduate student		X
Harrison	Richard	Mechanisms	Undergraduate student		X
Haycock	Ralph	Mechanical Testing	Professor		
Humphreys	Todd	ADS	Graduate student	X	X
Hunting	Steven	Electrical	Undergraduate student		
Isom	David	Integration and Testing	Undergraduate student		X
Jensen	John	Flight Computer	Graduate student	X	X
Kirkham	Greg	Gimbal	Graduate student	X	
Lee	Benjamin	Thermal	Undergraduate student		X
Lewis	Brian	Systems	Graduate student	X	X
Liang	Jinsong	ACS	Doctoral student	X	
Lin	Xi	Communications software	Graduate student	X	
Loertscher	Tyler	Power	Undergraduate student		X
Mathur	Rajat	Communications Antennas	Undergraduate student		X
Miller	Jon	Thermal	Undergraduate student		X
Mittal	Sanam	Real time software	Graduate student	X	X
Moffitt	Blake	Thermal	Graduate student		X
Nelsen	Joel	Software	Graduate student	X	
Nyman	Nichole	Software	Undergraduate student		X
Paulsen	Brandon	Thermal / Mechanical	Engineer		
Pulugundla	Srikanth	Documentation	Undergraduate student		
Quincieu	Joel Quincieu	Integration and Testing	Graduate student	X	
Redd	Frank	Principal Investigator	Professor		
Sanderson	Wayne	C&DH Hardware	Engineer		
Smith	Arron	I/O Hardware	Graduate student		X

Soma	Naveen	Software	Graduate student	X	
Sripruetkiat	Prapat	ADS Cameras	Graduate student	X	X
Stormont	Dan	Real time software	Graduate student		
Stuart	Michael	Flight Computer	Undergraduate student		X
Suisse	Brian	Mechanisms	Undergraduate student		
Swenson	Charles	Co-Principal Investigator	Professor		
Thorson	Darin	Science	Undergraduate student		X
Tulasiram	Sridhar	Structures	Undergraduate student		X
Vanhille	Ken	Power	Undergraduate student		X
Ward	Jeff	Science	Graduate student	X	
Whiting	Scott	Motor Control Electronics	Graduate student	X	
Wojcinska	Magdalena	Integration and Testing	Undergraduate student		X
Wooden	Jason	Mechanical / Booms	Engineer		
Young	Verl "Dino"	Mechanical Design	Undergraduate student		
Hazen	Amy	Wiring Harness	Undergraduate student		X

CHAPTER 1: INTRODUCTION

In recent years, many organizations including the U.S. Department of Defense (DoD) have started looking at small satellites. Technological advances have allowed these systems to be fabricated much less expensively and can be designed and completed in a short time. In addition, recent advances in microelectronics have allowed small spacecraft to perform missions that would have been impossible earlier. Small spacecraft are often used for basic research, in high risk programs and for educational involvement in space systems design. In addition to the DoD, many other groups within the National Aeronautics and Space Administration (NASA) have been working to include small spacecraft in their program goals. In 1999, the DoD and NASA decided to allocate funding for ten universities to begin designs on nanosatellites. These satellites would comprise 10 – 20 kg of mass and would be around the size of a small television. The focus of the design initiative was to have universities conduct creative low-cost space experiments to explore the military usefulness of nanosatellites in such areas as formation flying, enhanced communications, miniaturized sensors, attitude control, and maneuvering (Martin, Schlossberg, Mitola, Weidow, Peffer, Blomquist, Campbell, Hall, Hansen, Horan, Kitts, Redd, Reed, Spence and Twiggs 1999).

These spacecraft would be designed and fabricated by universities and then delivered to the Air Force Research Laboratory (AFRL) for integration with a new launch system to be used with the Space Shuttle being developed by Goddard Space Flight Center (GSFC). Utah State University (USU) was selected to design one of these spacecraft and was paired with the University of Washington (UW) and Virginia Polytechnic Institute – Virginia Tech (VT). These three universities would be a part of a constellation called the Ionospheric Observation Nanosatellite Formation (ION-F). These spacecraft would attempt to conduct experiments with upper atmospheric science and in formation flying.

The purpose of this thesis is to show how USU performed the systems engineering and safety engineering design allowing the spacecraft to evolve from a concept into a working system that fulfilled mission requirements and NASA safety requirements. The spacecraft and its preliminary, intermediate and final design characteristics are described. The limitations and design reasoning process that contributed to the evolution of the design are also described. The strategy used to make the spacecraft acceptable to NASA safety engineers is described as well. The methodology used in this design is expected to be useful to the small satellite design community.

USUSat Systems and Safety Background

The initial systems engineering work on USUSat was performed at SDL by early program management. The main engineers at this point were Pat Patterson and Brandon Paulson. These engineers had performed extensive systems, electrical and mechanical engineering work on projects at SDL. The Principal Investigator (PI) on the project was Dr. Frank Redd, but day to day work on the project was performed by a pair of Co-PI's, Dr. Rees Fullmer and Dr. Charles Swenson. Initial mass budgets and preliminary designs were completed by this group. The goal of this group was to act as advisors for students who would take over the detail design of spacecraft components.

A graduate student named Bryce Carpenter agreed to take over the systems engineering of USUSat in the fall of 1999 and the author of this thesis was assigned to be the safety manager. As such, the author's duties included ensuring that USUSat complied with applicable standards, preparing necessary paperwork for program reviews, and accompanying program management to safety and program reviews. In the spring of 2000, Bryce Carpenter left USU to accept a job and the author agreed to assume system engineering responsibilities in addition to safety engineering responsibilities for USUSat.

At this point in design, some redesign was necessary for some of USUSat's subsystems. Detailed design work had been completed for many components and while some worked well, some did not perform as desired. The main responsibilities would be to prepare all necessary documentation for program management and safety requirements, provide guidance for students who would complete the detail work in major subsystems, and finally, to help complete mechanical engineering design work in subsystems where other students were not available.

CHAPTER 2: SPACECRAFT SYSTEMS ENGINEERING

Development of Systems Engineering

A final definition given for spacecraft systems engineering is that it is "the art and science of developing an operable system capable of meeting mission requirements within imposed constraints including (but not restricted to) mass, cost, and schedule" (Griffin and French 1991).

USUSat Mission Definition

The US DoD has in recent years been investigating the feasibility of using small spacecraft in order to accomplish various military objectives including coordinated maneuvering, atmospheric science research, and educational involvement in research opportunities. In addition to the DoD, various groups within NASA have been working on projects with similar goals. In 1999, the DoD and NASA allocated funding and support through various subgroups for ten universities to begin working on nanosatellites. These groups included the Air Force Office of Scientific Research (AFOSR), the Defense Advanced Research Projects Agency (DARPA), AFRL, the Space Test Program (STP), and GSFC. The focus of the design initiative was to have universities conduct creative low-cost space experiments to explore the military usefulness of nanosatellites in such areas as formation flying, enhanced communications, miniaturized sensors, attitude control, and maneuvering (Martin et al. 1999).

Three of these universities, USU, UW, and VT were placed together into a group called the Ionospheric Observation Nanosatellite Formation (ION-F). USU was responsible for designing USUSat, UW for Dawgstar, and VT for Hokiesat. These three spacecraft were placed together to study the objectives described above, but also to see if three universities could successfully integrate design work over large distances. While some of the hardware on these spacecraft were to be identical, each university was ultimately responsible for the design on each satellite. These spacecraft were also intended to be a proof of concept flight for a new deployment system from the US Space Shuttle. As such, the spacecraft designs would be subject to NASA Safety requirements for design, fabrication, and documentation.

USUSat Formation Flying Objectives

One of the initial objectives of the University Nanosatellite Program (UNP) program was to demonstrate advanced formation flying objectives. This capability has been very influential to the design of the ION-F constellation. Formation flying objectives are of interest to the space community because they offer the possibilities for high level research for much lower cost. Formations of small spacecraft can perform research that would be prohibitively expensive if large spacecraft were to be used. Formations can be used to perform temporal and spatial research. These spacecraft could also be used to provide functionality that would otherwise require space based construction platforms.

The formation flying capabilities of the ION-F constellation will, in general, involve two main types of experiments. The first is described as leader-follower behavior. As a proof of concept design, USUSat has the ability to alter its ballistic coefficient in order to attempt

formation flying. This coefficient ranges both higher and lower than the ballistic coefficients of Dawgstar and Hokiesat. When USUSat is in a low drag configuration, Dawgstar and Hokiesat drop in orbit faster than USUSat and tend to separate from USUSat rather quickly. The opposite is true when USUSat is in a high drag configuration. One experiment to be tested is to see if the spacecraft can maintain stable distances between them. For this experiment, one spacecraft would remain stationary and the other would try to hold position relative to the first spacecraft. If USUSat was in a steady drag configuration, Dawgstar or Hokiesat would attempt to use thrusters to increase or decrease its relative velocity to match USUSat. If Dawgstar or Hokiesat is stationary, USUSat would attempt to rotate and modify its ballistic coefficient in order to match velocity. While USUSat has minimum and maximum ballistic coefficients, it can achieve any coefficient in between by using careful rotations.

A related experiment is to command separation. For example, if ION-F members could successfully maintain distances of, for example, 100 m apart; the next step would be to command them to move to 1000 m apart and hold distance and then return to 100 m difference.

The next step would be to see if all three members of the ION-F constellation could maintain distance simultaneously. Early tests would involve two spacecraft while the third was free to fly. Later tests would try to incorporate all three spacecraft. One would hold stationary while the other two would attempt to take up positions 100 m ahead and 100 m behind.

The second formation flying experiment deals with groundtracks. A formation could attempt to produce identical groundtracks as its spacecraft orbit. This would require the spacecraft to make small orbital inclination changes and to move out of their original track. In this case, one spacecraft would set a baseline groundtrack and a second would attempt to move out of track until its motion produced an identical groundtrack as the first. These experiments can also be extended to use all three spacecraft as well as just two. USUSat may be limited in such experiments since it is only able to produce minimal out of track forces that would allow it to adjust its orbital parameters. Dawgstar and Hokiesat would be the major spacecraft in this experiment.

USUSat Science Mission Objectives

The main science experimentation flown on the ION-F constellation is a pair of probe antennas that work the measure electron density and plasma frequency in the ionosphere. This research is of interest since the behavior of the ionosphere affects the propagation of radio signals. As our society depends more on satellite communications, navigation, and geolocation, better knowledge of ionospheric behavior is necessary in order to design better systems to accomplish these goals.

Some experiments have been conducted using sounding rocket payloads or using individual spacecraft. However, these tests do not allow experimenters to collect data on how the ionosphere evolves over time. Since ION-F has three spacecraft in a constellation, it is possible to take measurements of how ionospheric plasma evolves temporally and spatially. ION-F would be the first spacecraft constellation to make these systematic measurements as a group.

The science instrumentation that will be flown on the ION-F constellation was designed at SDL and is similar to instrumentation that has flown on previous payloads. USUSat has three main pieces of scientific equipment. The first is the deployable science boom. This boom is called a plasma impedance probe (PIP) and helps take measurements on plasma frequency,

electron density, and electron behavior in the ionosphere. A second piece of instrumentation is a small patch antenna called a DC probe (DCP). This patch helps provide relative electron density measurements. The final equipment is the electronics required to convert measurements into data.

The equipment on ION-F is intended to complete three major objectives. The first is to document the evolution of plasma structure and ionospheric irregularities. The second is to help determine the spectral characteristics of ionospheric plasma. The third is to help develop a global map of the distribution of plasma structures and irregularities.

USUSat Requirements Definition

ION-F was designed to be launched from the Shuttle Hitchhiker Experiment Launch System (SHELS). This system was designed as either a single or double sidewall launch system. In order to mate with this system, AFRL was responsible for designing a system called the Multiple Satellite Deployment System (MSDS). This system was developed in order to facilitate the deployment of multiple university payloads from the SHELS platform. ION-F would be combined onto one MSDS with a group called Three Corners Satellite (3CS). This group of spacecraft was designed by Arizona State University, Colorado State University, and New Mexico State University. The ION-F and 3CS combination meant that around 50 kg of mass was allotted to ION-F. Further, the ION-F group decided to partition 15 kg of mass to each spacecraft with the remaining 5 kg to be used as a design margin. In addition, the SHELS platform imposed limits upon stack geometry. Designers could use spacecraft that used a square or hexagonal footprint with height limits determined by the overall height of the spacecraft stacks. In order to be deployed, ION-F would have to be integrated together into a solid stack and then separate into three individual spacecraft in order to accomplish mission objectives. The spacecraft were designed to be joined and separated using a system called the Lightband developed by Planetary Systems Corporation (PSC).

USUSat was designed specifically to be launched as a secondary payload from the US Space Shuttle. Therefore, mission launch requirements were set by NASA safety engineers. In addition, mission profiles using the Space Shuttle are somewhat more limited than when using other launch systems. The shuttle is restricted in available launch azimuths as a safety precaution due to populated areas. In addition, the shuttle has altitude limitations that restrict payloads to LEO unless they carry secondary propulsion systems. With this in mind, the ION-F systems engineers requested some minimum orbital parameters. These parameters were developed using requirements for completing mission objectives and for communications.

The requirement for minimum altitude was derived from the required mission lifetime. The formation flying aspects of the ION-F mission were estimated to take around 60 days to complete so the minimum altitude was requested in order to attain this lifetime. The orbital decay was predicted using the Lifetime and HPOP functions included with Satellite Tool Kit (STK), a software suite designed to model spacecraft orbital performance. The results of the calculations performed by STK are shown in Figure 2. In order to ensure that ION-F has a 60 day lifetime, the minimum orbital altitude requested was 355 km.

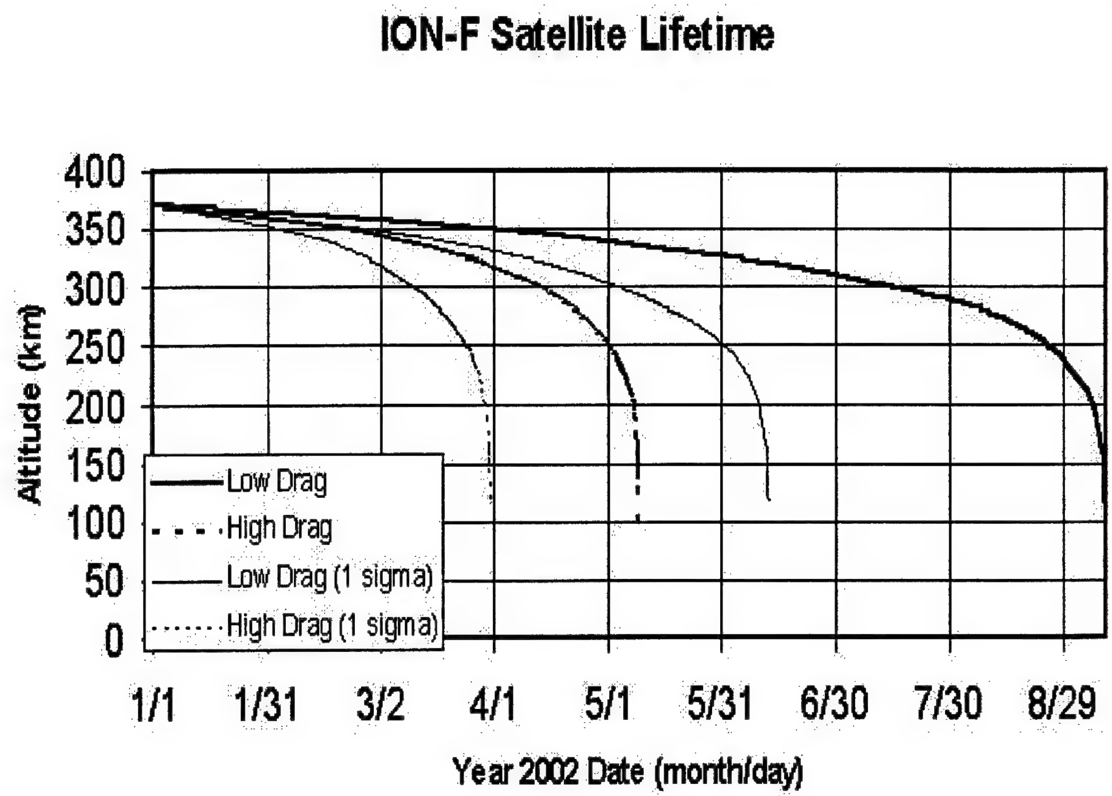


Figure 2. Orbital altitude vs. spacecraft lifetime.

In addition to mission lifetime requirements, orbital parameters were also set using communications requirements. As part of the ION-F mission, distributed scientific measurements of the upper atmosphere are to be measured. These measurements are to be taken rather rapidly and generate a significant amount of data. This data must be transmitted to the ground at a high rate. Ground stations for the ION-F constellation are not available in optimal locations for easy communication. Ground stations were planned to be placed at each of the three schools in the ION-F constellation. Unfortunately, these schools are all located at relatively high latitudes. Therefore, the constellation must be inserted into orbits with significant inclination for the spacecraft to communicate with the groundstations. Again, available communications time was simulated with STK in order to predict the minimum inclination required. Systems engineers determined that a minimum of 800 seconds of access time were required per day in order to successfully downlink all available data. As shown in Figure 3, the orbital inclination must be at least 36 degrees in order to provide adequate access time.

While these were the minimum requested parameters, ION-F was originally designed for an orbit similar to that of the ISS. ION-F's desired orbit was requested to have an altitude of 380 km and an inclination of 52 degrees. The calculations originally made assumed a launch date of January 2002. Later in the program it became apparent that due to ISS work, launch manifest would not available until much later. The minimum orbital parameters requested were based on increased solar activity in January 2002 and could be relaxed if desired by program management.

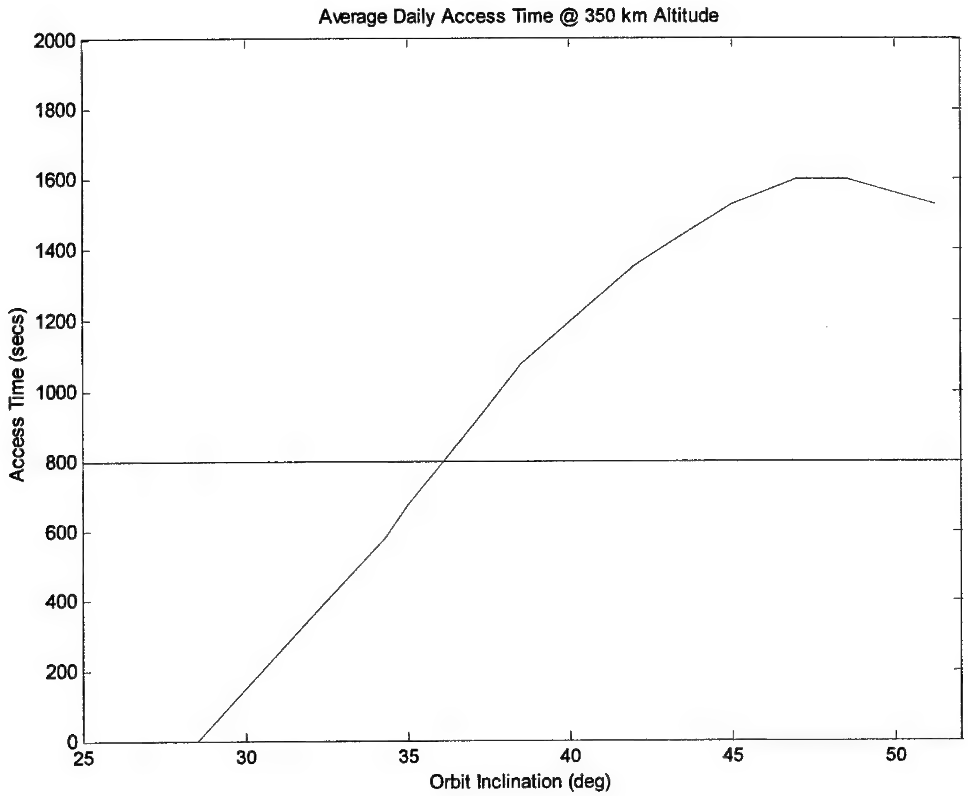


Figure 3. Daily access time vs. orbit inclination.

USUSat System Description and Concepts

ION-F was designed to be launched from the SHELS launch system on the Shuttle. The basic requirements for launch system compatibility came from two sources; the SHELS User's Guide (NASA, SSPP-SPEC-040 1999) and from the NASA safety group's requirements for payloads utilizing the Space Shuttle (NASA, NSTS 1700.7E 1989). These two documents outline the basic launch environment and safety requirements that must be fulfilled in order to use the SHELS system on the Shuttle. Some of the important requirements will be detailed further, but these documents contain too much information to completely summarize here.

In order to accomplish the mission objectives discussed earlier, ION-F systems engineers identified the following needs for the spacecraft in the ION-F constellation. In order for these spacecraft to autonomously perform coordinated maneuvers, the spacecraft needed some form of interspacecraft communication. In addition, propulsion systems or some way of altering spacecraft velocity would be needed in order to perform these coordinated maneuvers. Precise maneuvers would require precise attitude determination and control systems. Finally the science payloads would need to have antennas that could be deployed away from the spacecraft in order to make readings on undisturbed portions of the ionosphere. As stated before, these systems would have to be packaged into 15 kg of mass, not an insignificant challenge.

With these necessary subsystems and mission parameters, USUSat engineers began to build a conceptual design of the spacecraft. Originally, designers wished to keep the spacecraft

as simple as possible. The structure was originally planned to be a perfect hexagon in shape with a major diameter of 19.75" and a height of 5.5". The spacecraft structure was designed to be fabricated from 6061 sheet aluminum with machined aluminum corner struts. A small flight computer would interact with system sensors including two deployable booms. One boom would act as an antenna to take atmospheric measurements while the other would contain a magnetometer to be used for attitude determination. Horizon and sun sensors as well as rate gyros would also be used in order to provide accurate attitude knowledge. Finally a GPS receiver would be used to provide accurate locations. Control actuation would be done through the use of a new technology, gimbaled permanent magnets. The spacecraft would use a technique called differential drag in order to adjust its velocity relative to the other spacecraft in the ION-F constellation. Designers chose to use body mounted solar cells rather than deployable panels and selected Nickel Metal Hydride (NiMH) batteries for use due to their cycle life and depth of discharge characteristics as well as for the increased power density that they exhibited. The communications subsystem contained a receiver and transmitter, as well as the GPS receiver and a crosslink transceiver. Finally, designers believed that thermal limits could be maintained through the use of passive coatings and a few Kapton strip heaters where necessary. The equipment was to be kept simple and small. Figure 4 shows an early conceptual idea for internal component placement.

Using these ideas, an early mass and power budget was developed for the system. This budget was developed with inputs from several people working on the key subsystems for the project. This preliminary budget is shown in Table 1. This table shows the projected masses, peak budgets and orbit average power (OAP) budgets for the spacecraft. It is interesting to note that the largest percentages of mass were allocated to the magnetic control system and to the power subsystem.

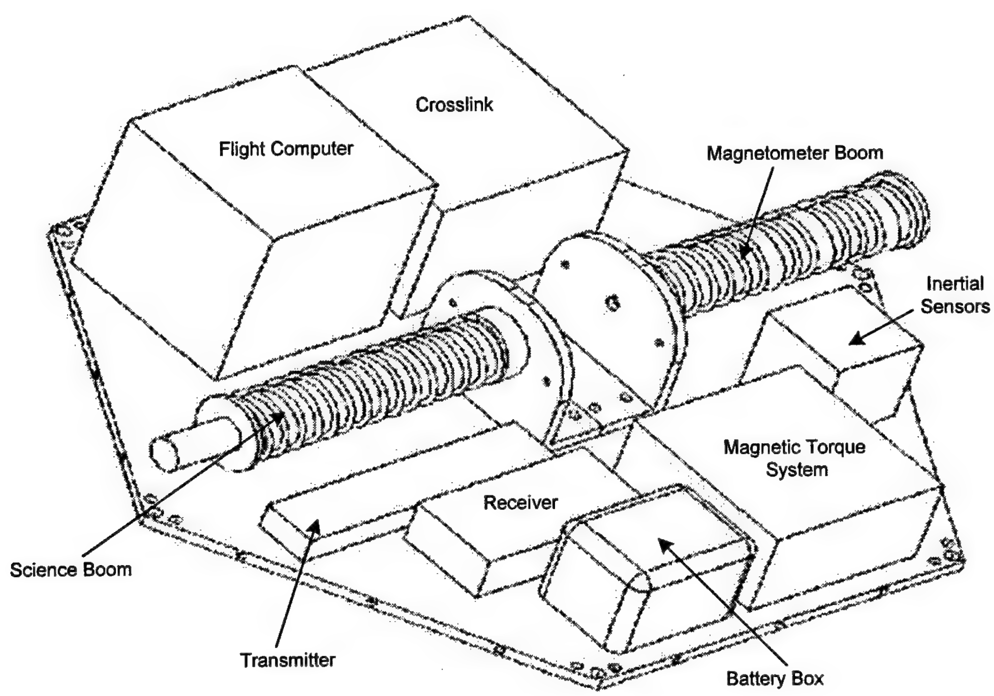


Figure 4. Preliminary component placement.

In order to explain this result it may be useful to explain more about these subsystems and why their preliminary design included this much mass. As stated above, each spacecraft was designed to participate in formation flying objectives. USUSat chose not to use propulsion systems that the other two spacecraft in the formation used. Instead USUSat chose to use drag modulation. While there is very little atmosphere at USUSat's design altitude, there is some and it can have significant effects over time. In fact, it is this drag that causes satellite orbits to slowly decay and eventually reenter the earth's atmosphere and burn up. Spacecraft or rocket designers use a parameter called the ballistic coefficient to measure the magnitude of the atmospheric drag effect. The ballistic coefficient is essentially the ratio of surface area to mass. As noted above USUSat has a very large diameter in comparison to its height. Dawgstar and Hokiesat, the other spacecraft in the ION-F constellation have heights that are much closer to their diameters, between the extremes presented by USUSat. The effect of this geometry is that Dawgstar and Hokiesat have nearly constant ballistic coefficients while USUSat can dramatically change its coefficient by altering which face is aligned with its velocity vector or ram direction. Therefore USUSat can in effect speed up or slow down relative to Dawgstar or Hokiesat by altering its alignment.

In order to change its alignment in this manner, USUSat needs precise three axis control capabilities. Two standard ways are to use propulsion systems or to use torque coils, in effect, large electromagnets, that interact with the earth's magnetic field in order to orient the spacecraft as desired. These systems generally require large amounts of electrical power, power that is in short supply for small spacecraft. So an experiment was undertaken to see if an alternate form of magnetic control could be found that would consume less power. Permanent magnets would be gimbaled or rotated using stepper motors in order to align their magnetic vectors in desired directions in order to rotate the spacecraft to point as necessary. Therefore, a large mass budget was allocated to designing the magnets and the gimbal that would orient them as required. Since such an experiment had never flown before, a large margin was allocated to allow designers flexibility in completing their design.

The second large mass allocation was made for the power subsystem. While a large amount of mass was set aside for cabling, this is somewhat standard. The battery packs also received a large mass allocation. For safety reasons that will be discussed further in later sections, ION-F systems engineers were asked to use deployable systems only where absolutely necessary. For this reason, designers chose to use a series of body mounted solar panels rather than to use deployable panels. Due to the small size of the side panels and the Lightband interface ring on the lower panel, the majority of the solar cells were placed onto the large upper panel. However, formation flying considerations required that USUSat fly in certain orientations in order to maintain desired ballistic coefficients. This could result in a series of orbits in which very little power would be generated if the large panel could not be oriented with the sun. Therefore, a large amount of mass was allocated to the batteries so that sufficient storage capacity would be available for those orbits in which very little power was generated by the solar arrays.

Table 1. Preliminary Mass and Power Budgets

Subsystem	Component	Mass (g)	Peak Power (W)	OAP Power (W)
Structures	Base Plate	454.0	0.00	0.00
	Top Plate	454.0	0.00	0.00
	Side Panels	454.0	0.00	0.00
	Lightband	680.0	0.00	0.00
	Fasteners	181.0	0.00	0.00
Mechanisms	Magnets	985.0	0.00	0.00
	Stepper Motors	181.0	5.00	0.10
	Gimbal Structure	680.0	0.00	0.00
	Electron Probe Boom	227.0	0.00	0.00
	Magnetometer Boom	272.0	0.00	0.00
	Deployment Actuators	91.0	0.00	0.00
Power	Power Regulation	45.0	1.00	1.00
	Solar Cells	455.0	0.00	0.00
	Batteries	2725.0	0.00	0.00
	Cabling	905.0	0.00	0.00
Thermal	Kapton Strip Heaters	136.0	2.00	0.05
	Temp. Monitors	5.0	0.10	0.01
	Thermostats	50.0	0.00	0.00
Communications	GPS Receiver	680.0	0.00	0.00
	S-Band Transmitter	0.0	8.00	0.05
	Receiver	282.0	1.00	1.00
	Beacon Transmitter	0.0	0.00	0.00
	Link Matching Circuits	0.0	0.00	0.00
	Data Formatter	907.0	0.10	0.10
	Crosslink / GPS	454.0	2.50	2.50
	Antennas	0.0	0.00	0.00
C&DH	Flight Computer	30.0	1.05	0.85
	Data Buffer	55.0	0.23	0.03
	Shielding	90.0	0.00	0.00
ADCS	CMOS Camera	400.0	1.50	0.50
	Magnetometer	50.0	0.20	0.20
	Sun Sensor	600.0	0.20	0.10
	Camera Electronics	0.0	0.00	0.00
	Control Electronics	0.0	0.40	0.10
	Torquer Coils	181.0	6.00	0.05
	Rate Sensors	91.0	2.00	1.00
Science	Plasma Probe	227.0	1.50	1.50
	GPS Signal Strength	227.0	0.00	0.00
	Total	**13692.0**	**32.78**	**9.14**

USUSat Fabrication and Test

In order to ensure proper workmanship on manufactured items, USUSat has either procured certificates of conformance for purchased items or manufactured parts within SDL's quality assurance system. SDL maintained a list of qualified engineers who were required to approve all designs that were submitted for manufacture. This list of engineers all had to approve purchasing decisions. Electronic parts all conformed to those listed on GSFC's Approved Manufacturers and Parts List. Designs were also presented for review to the other members of the ION-F constellation as well as to program management at AFRL.

Assembly of electronic parts and components was completed in a class 10,000 clean room at SDL. Electro-Static Discharge (ESD) controls were implemented in handling parts. All parts were checked into SDL's tracking system thus allowing parts to be traceable for handling and assembly procedures.

The testing regime for USUSat and ION-F has not been fully completed yet but test plans are being prepared by Jöel Quincieu at SDL. The test regimen is designed to meet requirements set by program management and by NASA safety as well as to ensure that the satellite will function as designed. Planned tests include sine sweep, sine burst and random vibration tests, mass properties determination, electric continuity tests in the inhibit system, and powered vibration.

USUSat Design Philosophy

USUSat was designed to be a secondary payload for use on the Shuttle, but it was also intended to be designed by students. It was to be developed on a small budget and was intended to be a high risk payload. As such, the spacecraft was originally intended to be as simple as possible. Designers wanted to use a sheet aluminum structure and pick COTS electronics wherever possible. Most systems would have no form of backup and the spacecraft would be designed to operate autonomously wherever to possible to reduce the support staff necessary.

Since designers had a small budget to work with, they tried to recruit a small number of graduate students who could use the work on the project as part of their thesis or dissertation. Undergraduate students were recruited by allowing them to use the project for their engineering design course requirements. Three USU faculty members would act as the project PI's and provide guidance to students. In addition, SDL was located close to campus and managers felt they could draw on SDL expertise and facilities if needed.

The spacecraft would be launched from the Space Shuttle and so managers knew that some paperwork would be inevitable. The project was designed to present the fewest number of hazards possible in order to minimize the impact that safety would have on the project. Designers worked to eliminate stored energy sources wherever possible. Batteries were selected to be as benign as possible, using technology that had flown on previous missions. In addition, the spacecraft was designed to operate under a condition called the "unpowered bus exception". This exception deals with power flow within the spacecraft and the methods required for monitoring this flow. This exception will be discussed in greater detail in Chapter 4, but, in essence, the spacecraft was designed so that no power would flow in the spacecraft until after the

Shuttle had landed, ensuring that the spacecraft could not activate any hazardous functions while in the payload bay.

USUSat Design Characteristics

Internal Layout and Design

Using the basic mission concept and the geometry available, USUSat designers began to add design detail to the preliminary design for USUSat. Preliminary internal component placement was shown in Figure 4 and a preliminary mass and power budget were shown in Table 1. The interior volume and external area of USUSat had to accommodate all of the subsystems necessary for USUSat's operation. Due to USUSat's small height, fitting all the necessary components would be a significant challenge for designers. This process was further complicated by the fact that several of the parts were being designed by institutions other than USU and that the designs were concurrent. Often changes in one component or another would require internal components to be moved to allow for sufficient room not only for the boxes that would hold the electronics gear, but also for the cables and connectors that transmitted signals throughout the spacecraft.

The first internal layout that was completed is shown in Figure 4. In this design, the major systems were the deployable booms. These systems took up the most space within the center of the spacecraft. The rest of the electronics would be placed around the booms in boxes small enough to allow for cabling to pass in between.

The next layout that was completed was done to integrate the new requirements for the USUSat Common Electronics Enclosure (CEE). The enclosure designed to house the electronics would now be packed into a long, low box with connectors that routed wiring out through the top. The inertial rate gyros were hung from the side of this box and so the entire box was rather long. Also causing concern were the magnetic gimbal and the crosslink chassis. The magnetic gimbal required around 5.5" of internal space to allow for free rotation. USUSat has 5.5" total internal height and no other equipment would be allowed to occupy any position within the rotational volume. The crosslink chassis was an electronics enclosure design that was similar to the ION-F CEE. These components needed to be moved toward the center of the spacecraft in order to fit within the volume allowed. This forced the booms to be redesigned to occupy the space over the top of the electronics enclosure and crosslink chassis. The results of this internal redesign are shown in Figure 6.

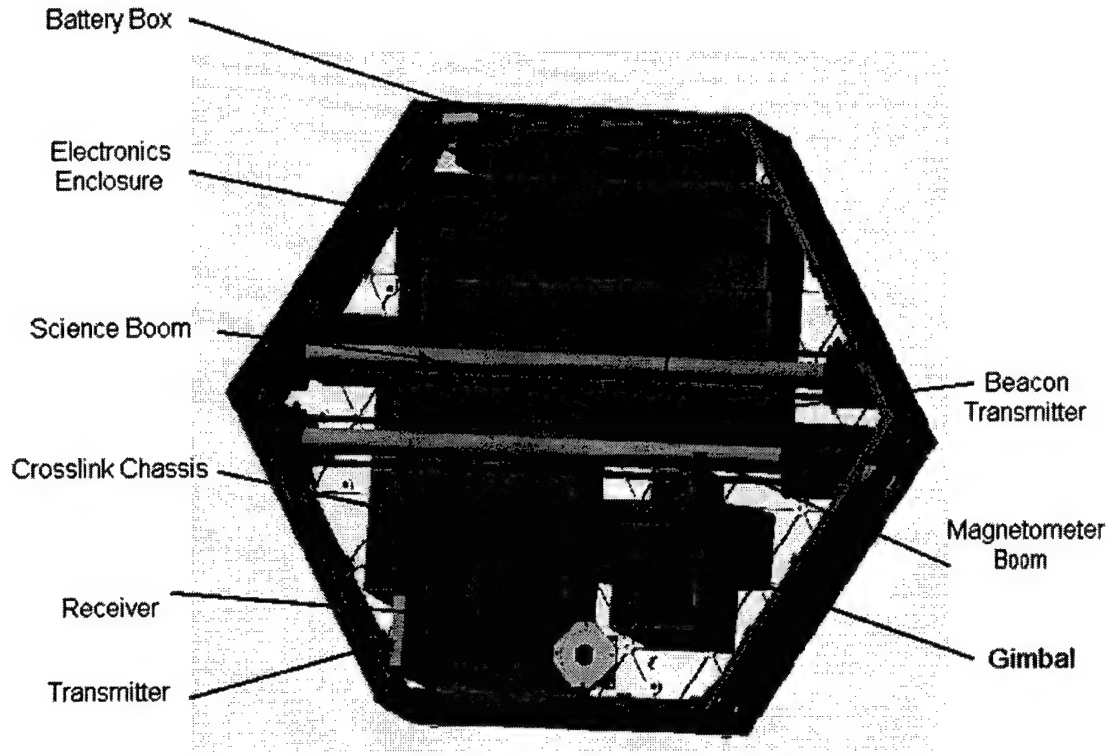


Figure 6. USUSat internal layout in intermediate stages.

The USUSat design would undergo one more major internal revision. As shown in Figure 6, the spacecraft receiver and transmitter were placed near the bottom of the spacecraft in order to minimize the cable length that was needed to interface with antennas. This also forced the magnetic gimbal to be located near the bottom of the spacecraft. Communications engineers felt that the presence of the permanent magnets would interfere with reception and transmission of signals so they requested that the gimbal be moved. In addition, the extra communications gear that was associated with the crosslink system was added around this time. Figure 7 shows the nearly finalized internal configuration of USUSat.

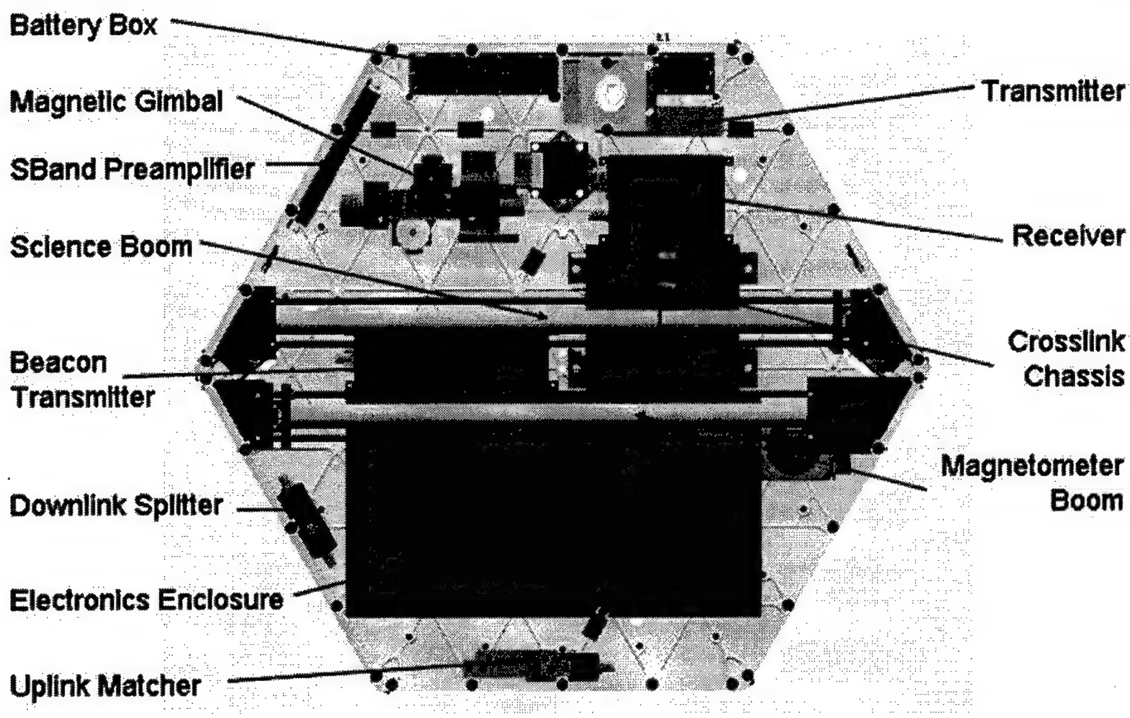


Figure 7. USUSat internal configuration in final stages.

In this configuration, the major internal components have been rotated 180°. The computer enclosure is now located near the bottom of the spacecraft and the gimbal and communications gear have been shifted toward the top. In addition, the link splitting and matching circuits are visible as is one of the preamplifiers for the crosslink system. For the final design configuration, another preamplifier was located near the lower right panel, a switch and isolator were placed on the bottom panel, and the downlink transmitter was placed on the upper right panel.

While it was finally possible to fit all the required components into USUSat, for some time it appeared that all these components would not fit. The attitude determination and control (ADCS) system had originally requested eight cameras instead of four, as well as inertial rate gyros. The proposed deployable antennas would have required significant internal volume to accommodate pin pullers and tensioning mechanisms. The elimination of these systems from the USUSat design allowed a successful internal layout to be completed.

External Layout and Design

In contrast to the internal layout, there was only one area of the external design that proposed real challenges. Balancing solar cells and antennas took the most effort. The solar cells were laid out according to guidelines received from the cell manufacturer, Tecstar. Cells had 0.030" spacing between each cell and 0.2" spacing from the cells to the panel edges. Solar cell arrays and antennas both had to be placed so that their cabling would not interfere with internal equipment.

The largest problem arose during the design of the USUSat bottom panel. The bottom panel has the Lightband separation system and so there is a ring on the outside of the panel that is designated as a stayout zone. However, objects could be placed within the center of the ring. Power systems engineers wanted to place two strings of solar cells there in order to collect incoming energy. The Lightband ring would protrude 1.122" from the face of the USUSat nadir panel, thus shadowing the cells if the incoming light was at a large angle. Some 0.75" aluminum honeycomb was obtained to raise the cells off the surface of the panel. After this, the deployable antennas were changed into an array of patches and a copper ring. One patch antenna and the copper ring had to fit onto the bottom panel in the center of the panel. The honeycomb was cut into small pieces and arranged in a diamond outside the copper ring and inside the Lightband stayout zone. Figure 8 shows the arrangement of components on the nadir panel.

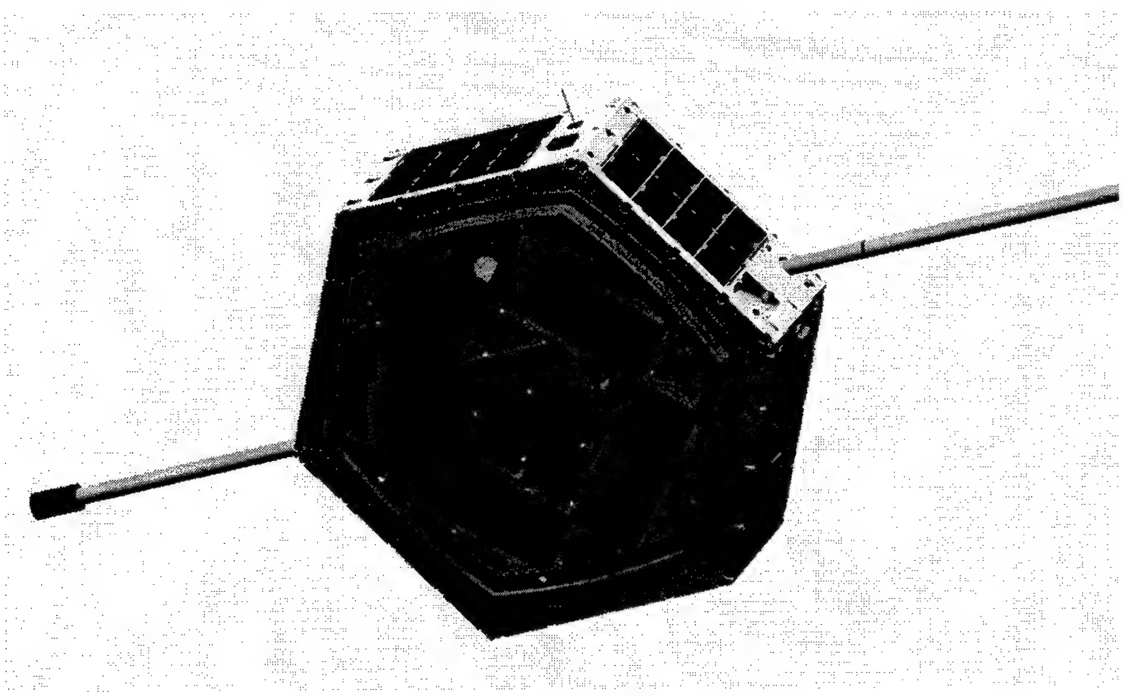


Figure 8. External equipment on nadir and side panels.

The top panel of USUSat was considerably easier. Five strings of solar cells were placed as well as a small location for an auxiliary port and fastener locations for stack lifting hardware. Figure 9 shows the arrangement of these components on the zenith panel. Three side panels also received one string of solar cells each. The booms were designed to be attached to two panels at each side and to eject from the upper left and lower panels when viewed from above.

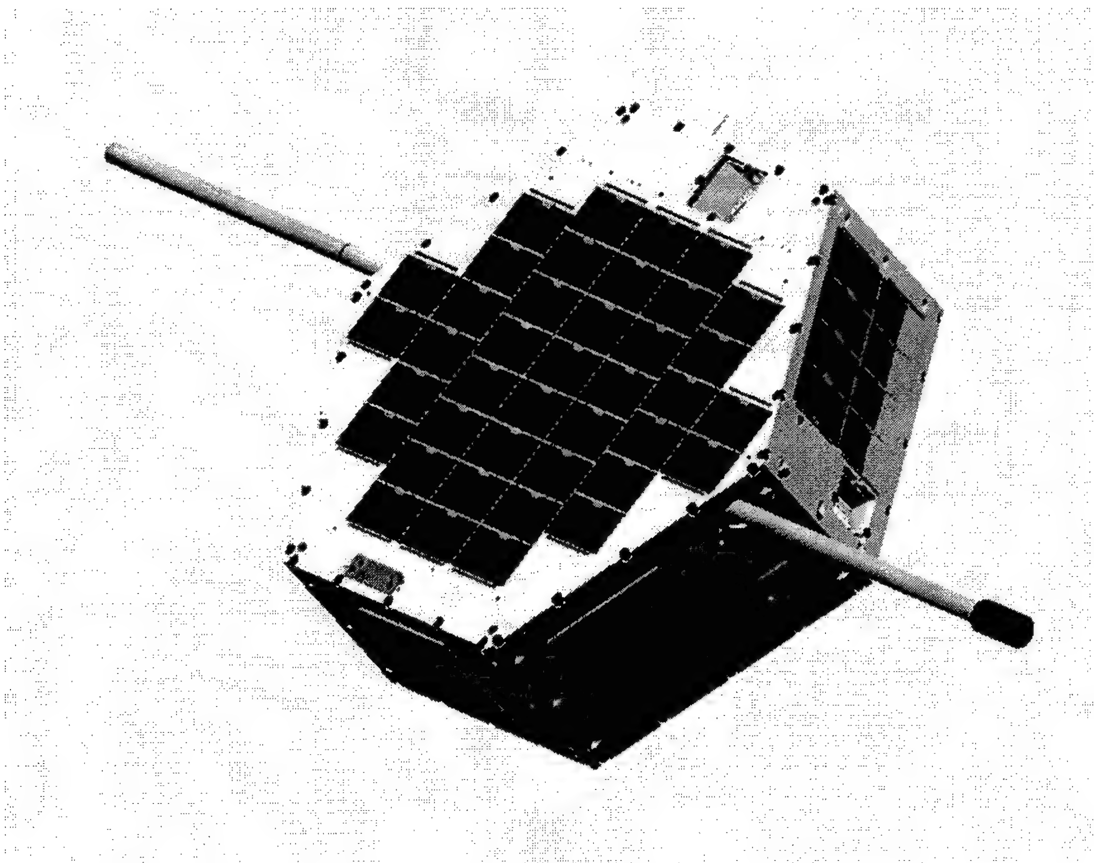


Figure 9. External equipment on zenith and side panels.

A functional block diagram (FBD) showing the structural panels, their connection points and the components attached to each, both internal and external, is shown in Figure 10.

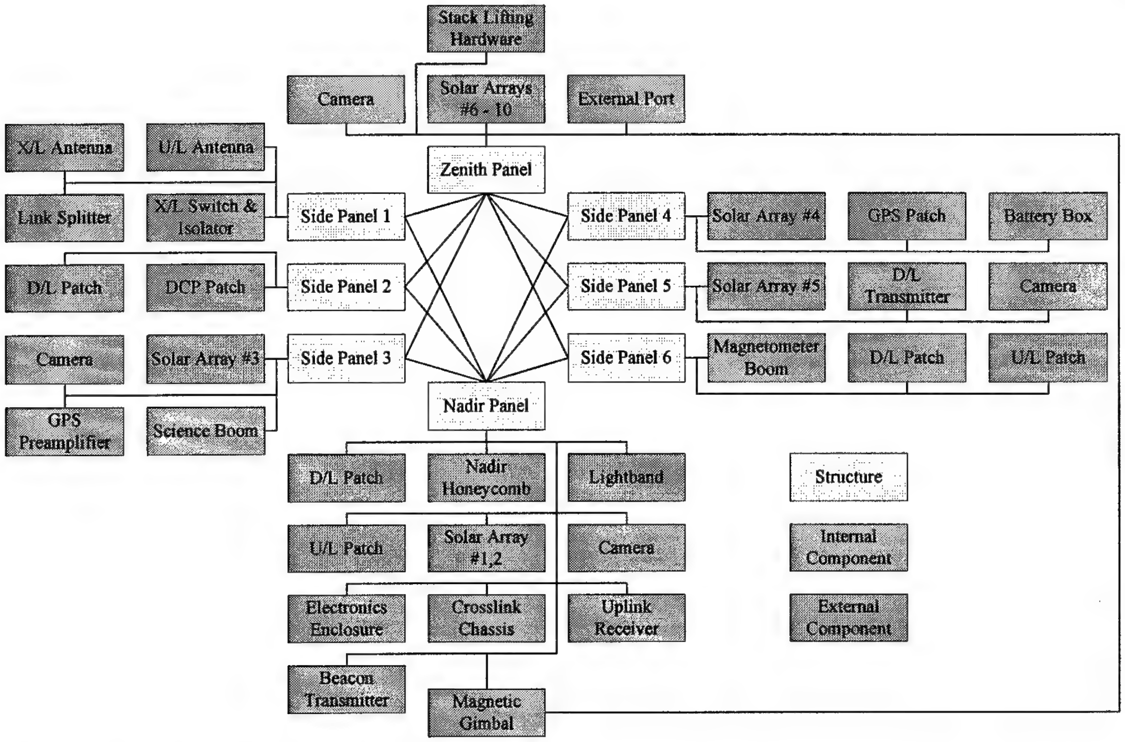


Figure 10. FBD of structural attachments.

Analysis of Design

One measure of how well designers have completed their job is to compare final results with preliminary estimates. This comparison can be useful in several ways. It can help reveal the extent to which a design has been optimized. Systems that use existing technology should fall relatively close to their original estimated levels while new technologies or systems often have a great degree of variability in their final characteristics. Table 1 previously documented the original specifications for USUSat. Table 2 shows how closely the actual mass came to the predicted mass while Table 3 shows how the power budget evolved.

Table 2. Actual System Mass Compared to Preliminary Budget

Subsystem	Component	Predicted Mass (g)	Actual Mass (g)
Structures	Base Plate	454.0	725.0
	Top Plate	454.0	625.0
	Side Panels	454.0	1344.0
	Lightband	680.0	811.0
	Fasteners	181.0	580.0
Mechanisms	Magnets	985.0	152.0
	Stepper Motors	181.0	318.0
	Gimbal Structure	680.0	370.0
	Electron Probe Boom	227.0	338.0
	Magnetometer Boom	272.0	358.0
	Deployment Actuators	91.0	87.0
Power	Power Regulation	45.0	819.0
	Solar Cells	455.0	398.0
	Batteries	2725.0	1358.0
	Cabling	905.0	2750.0
Thermal	Kapton Strip Heaters	136.0	10.0
	Temp. Monitors	5.0	50.0
	Thermostats	50.0	20.0
Communications	GPS Receiver	680.0	0.0
	S-Band Transmitter	0.0	203.0
	Receiver	282.0	232.0
	Beacon Transmitter	0.0	383.0
	Link Matching Circuits	0.0	178.0
	Data Formatter	907.0	172.0
	Crosslink / GPS	454.0	1292.0
	Antennas	0.0	349.0
C&DH	Flight Computer	30.0	105.0
	Data Buffer	55.0	158.0
	Shielding	90.0	779.0
ADCS	CMOS Camera	400.0	372.0
	Magnetometer	50.0	20.0
	Sun Sensor	600.0	0.0
	Camera Electronics	0.0	167.0
	Control Electronics	0.0	244.0
	Torquer Coils	181.0	0.0
	Rate Sensors	91.0	0.0
Science	Plasma Probe	227.0	20.0
	GPS Signal Strength	227.0	216.0
	Total	**13692.0**	**16003.0**

One will notice that the final design is roughly 2.3 kg more massive than preliminary estimates had indicated. While there is no answer that one can directly identify, there are a number of small contributors that sum up to explain the increase. The structure system was originally designed to be fabricated of sheet aluminum that would be joined in the corners using

aluminum struts. However, in order to make this design stiff enough to meet launch vibration requirements, the structural mass would have well exceeded its budget. The structure was finally designed using aluminum isogrid and while it exceeds its original budget, the isogrid is still lighter than solid plate. It appears that initial estimates were unrealistic in this case. In addition, one can see that the program requirements began to creep during the design. Additional elements were added to the design and mass increased to accommodate these changes. In addition, NASA safety engineers and APL communications engineers asked for changes that added mass to the design. These changes were unforeseen originally. One final reason for the increase is that the original estimates were made by SDL engineers who were experienced in optimizing a design. The design was subsequently turned over to USU students who lacked the experience to completely optimize the equipment.

Two last notes may also be enlightening. While the original budget was for 13.7 kg, USUSat actually had an allowable mass budget of 15 kg. While the current mass is still greater than 15 kg, the difference is much smaller. Finally, USUSat engineers have included around 2.75 kg of mass for cabling. The extra equipment growth resulted in additional internal wiring. This estimate also became a form of margin. Engineers expect around 750 – 850 grams to be returned leaving the actual spacecraft mass only around 200 g over budget. These factors are discussed in greater detail in Chapter 3 where more detailed descriptions of USUSat's subsystems are available.

Table 3. Actual Power Consumption vs. Preliminary Budget

Component	Est. Peak Power (W)	Est. OAP Power (W)	Act. Peak Power (W)	Act. OAP Power (W)
Base Plate	0.00	0.00	0.00	0.00
Top Plate	0.00	0.00	0.00	0.00
Side Panels	0.00	0.00	0.00	0.00
Lightband	0.00	0.00	60.00	0.00
Fasteners	0.00	0.00	0.00	0.00
Magnets	0.00	0.00	0.00	0.00
Stepper Motors	5.00	0.10	4.16	0.16
Gimbal Structure	0.00	0.00	0.00	0.00
Electron Probe Boom	0.00	0.00	0.00	0.00
Magnetometer Boom	0.00	0.00	0.00	0.00
Deployment Actuators	0.00	0.00	28.00	0.00
Power Regulation	1.00	1.00	0.00	0.00
Solar Cells	0.00	0.00	0.00	0.00
Batteries	0.00	0.00	0.00	0.00
Cabling	0.00	0.00	0.00	0.00
Kapton Strip Heaters	2.00	0.05	2.80	0.18
Temp. Monitors	0.10	0.01	0.00	0.00
Thermostats	0.00	0.00	0.00	0.00
GPS Receiver	0.00	0.00	0.00	0.00
S-Band Transmitter	8.00	0.05	28.00	0.34
Receiver	1.00	1.00	2.00	1.00
Beacon Transmitter	0.00	0.00	10.80	0.17
Link Match Circuits	0.00	0.00	0.00	0.00
Data Formatter	0.10	0.10	0.88	0.38

Crosslink / GPS	2.50	2.50	10.20	7.18
Antennas	0.00	0.00	0.00	0.00
Flight Computer	1.05	0.85	2.20	1.54
Data Buffer	0.23	0.03	0.32	0.18
Shielding	0.00	0.00	0.00	0.00
CMOS Camera	1.50	0.50	1.20	1.00
Magnetometer	0.20	0.20	0.25	0.20
Sun Sensor	0.20	0.10	0.00	0.00
Camera Electronics	0.00	0.00	0.00	0.00
Control Electronics	0.40	0.10	0.91	0.41
Torquer Coils	6.00	0.05	0.00	0.00
Rate Sensors	2.00	1.00	0.00	0.00
Plasma Probe	1.50	1.50	2.10	1.50
GPS Signal Strength	0.00	0.00	0.00	0.00
Total	**32.78**	**9.14**	**153.80**	**14.24**

From Table 3, one can see that the peak power consumption was significantly higher than originally estimated. The main reason for this difference is that the original estimates did not include a peak usage for the separation system and deployment actuators for the deployable booms. In addition, the communications equipment had much higher power consumption rates than originally estimated. The average power consumption is much closer to the original estimate. The main difference is in the GPS – crosslink system. This system was designed outside of the ION-F group and was designed for systems that have significantly higher power generation rates than USUSat. Communications engineers are working to find an acceptable method of power cycling the crosslink, such as transmitting only at given intervals, which will lower the power consumption. If this fails, USUSat will have to spend more time in a sun-pointing mode thus forcing it to spend less time meeting its formation flying objectives.

In addition to comparing original estimates with final specifications, it is often useful to compare a design to other contemporary spacecraft. While no two spacecraft will be the same, designers can often tell whether they have allocated too much mass or power to certain subsystems or whether they have been able to complete a design that can help advance the technology used in spacecraft design. In order to make comparisons, it is necessary to have the data from other spacecraft. Wertz and Larson (1999) give distributions of mass for a few selected spacecraft. Heffernan (1987) also gives mass distributions of the mass of selected Scout class spacecraft. Scout class spacecraft are small spacecraft since the mass injection capability of the Scout launch system is small. The results of these surveys are shown in Table 4 compared to the data from USUSat.

Table 4. Comparison of USUSat Mass Distribution vs. Other Spacecraft

Subsystem	USUSat: Mass (kg)	USUSat: Percentage	USUSat: Reclassified (kg)	USUSat: Reclass. Pct.
Payload	0.24	1.47 %	0.64	3.98 %
Structure	5.73	35.79 %	4.09	25.53 %
Thermal	0.08	0.50 %	0.08	0.50 %
Power	5.33	33.28 %	5.33	33.28 %
Communications	2.81	17.55 %	2.81	17.55%
C&DH	1.04	6.51 %	1.04	6.51%

ADCS	0.78	4.89 %	2.02	12.65 %
Propulsion	0.00	0.00 %	0.00	0.00 %
	USUSat: Reclass. Pct.	**Percentages: Wertz and Larson - Large**	**Percentages: Lightsats**	**Percentages: Heffernan – Scout Class**
Payload	3.98 %	26.70 %	24.40 %	14.62 %
Structure	25.53 %	21.70 %	22.70 %	19.79 %
Thermal	0.50 %	3.40 %	1.70 %	2.82 %
Power	33.28 %	27.90 %	24.60 %	23.12 %
Communications	17.55%	3.25 %	6.35 %	6.45 %
C&DH	6.51%	3.25 %	6.35 %	6.94 %
ADCS	12.65 %	8.00 %	11.30 %	15.34 %
Propulsion	0.00 %	3.70 %	2.70 %	10.92 %

In Table 4, the mass distribution of USUSat is compared with others reported in applicable literature. Wertz and Larson (1999) give mass distributions for a range of different spacecraft, mostly large spacecraft. The overall distribution that they report is shown in column 5. In addition, their reported mass distributions for lightsats or small satellites are shown in column 6. These distributions should be more applicable to USUSat since these spacecraft will have had to make some of the same systems engineering level decisions on design that USUSat did. In column 7, Heffernan (1987) reports on some Scout class spacecraft, again small satellites that will be comparable to USUSat.

One can notice in Table 4 that the largest subsystems, by mass allocation, on USUSat are the structure and power subsystems. In comparing these to reported distributions, the first note is that USUSat seems to have a much lower percentage of mass allocated to its payload than most spacecraft. It also has a much larger allocation for communications equipment than most. It should be pointed out that some of the science equipment was classified as mechanisms which were included with the structure subsystem for this comparison. In addition, the magnetic gimbal and magnetometer boom were included with the mechanisms. If the mass on USUSat is reclassified with the science boom classified as payload and gimbal and magnetometer boom as ADCS, the subsystem mass distributions become much closer to those reported for other spacecraft.

This reclassification could be taken even further. The crosslink communications system was included for formation flying purposes as was the magnetic gimbal. These systems could be classified as payload, in which case the USUSat mass distribution would be even closer to those reported.

One last factor to be considered is that small spacecraft showed the greatest deviation from average designs. Wertz and Larson also reported the standard deviation as well as the average values for small satellites. Payload and structures for small satellites had standard deviations of 9.4% and 7.7% respectively, while larger satellites had 4.2% and 3.3% respectively. USUSat's mass distribution then seems to fit very well within average values for small satellites and would lead to the assumption that the design has been reasonably optimized.

Program Management

Another aspect that must be considered when working on a spacecraft design project is the methods that will be use to manage information, personnel and resources within the project. The ION-F mission was conceived and designed by engineers; consequently much of the management structure was never formally defined but evolved as the project progressed.

Program Interaction and Information Flow

Each school within the ION-F constellation had a separate principle investigator (PI). At USU, Dr. Frank Redd was originally designated as the PI for USUSat with Dr. Rees Fullmer and Dr. Charles Swenson as advisors. Control of the project was then transferred to these Drs. Fullmer and Swenson as co-PI's for most of the project. Late in the project, Dr. Swenson came to take over as sole PI. Each school also had a lead systems engineer. Ideally, each school was also supposed to have safety and test leads as well with several subsystems. Each subsystem team would have a lead and this person would interface with the system engineer. In this way, the systems engineer could stay informed about progress on the project and convey requirements and decisions to the subsystems. The safety lead would be involved to make sure that designs would be satisfactory and to help produce NASA's required paperwork. The test and integration lead was to help with manufacturing and assembly issues.

ION-F also had an overall systems lead to which each school would report. This lead was responsible for interfacing with program management at AFRL. There were also ION-F safety and test leads who reported to their AFRL counterparts. This management structure is shown in Figure 11.

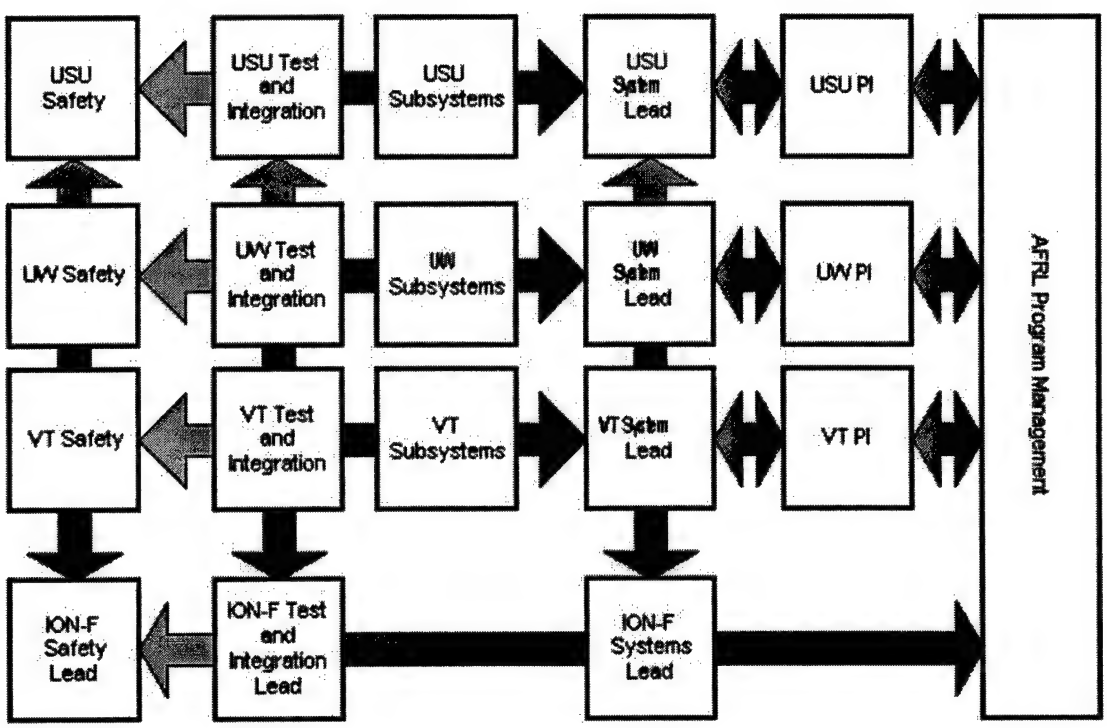


Figure 11. ION-F management structure.

This structure worked to distribute information within the ION-F group, but additional structure was needed to convey requirements and launch system information from the ION-F customers to the design groups. It was also necessary to communicate the design characteristics back to the customers. In this case, the customers are the AFRL and GSFC. They had contracted with the universities to provide the spacecraft and had offered their services in integrating the payloads and helping push necessary paperwork through the NASA system. This structure is shown in Figure 12.

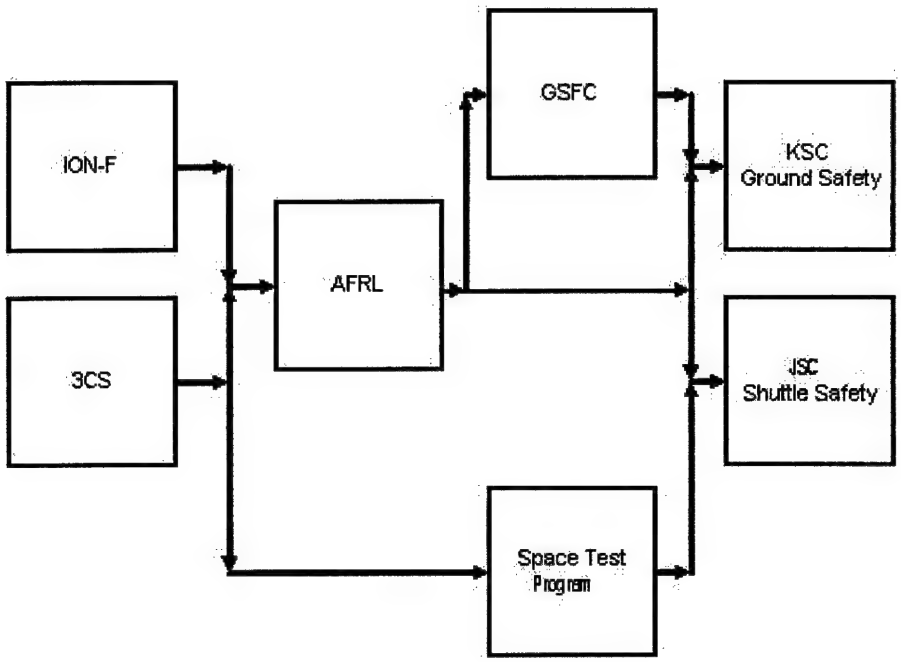


Figure 12. ION-F information flow structure.

The ION-F constellation was paired with a second constellation, 3CS, and they were designed to be launched together on one MSDS. This group would be called the University Nanosatellite 2 (UN2) payload. UN1 was composed of spacecraft from Stanford and Santa Clara Universities. As such, ION-F was responsible for reporting design progress to AFRL which designed the MSDS and was responsible for UN2. AFRL had to deliver the UN2 payload to GSFC where it would be integrated with the SHELS launch system. As such, AFRL was responsible for providing all necessary safety information to GSFC managers. These two organizations would then be responsible for delivering hardware and safety information to KSC and JSC engineers. KSC engineers were responsible for ensuring that the hazards during ground operations were controlled and JSC engineers were responsible for controlling hazards during space flight.

Since the UN2 program was student designed, Space Test Program (STP) engineers offered their technical expertise. STP representatives were present during program reviews and offered suggestions that would help alleviate concerns about the design. In addition, they made

themselves available for students to contact when they had questions about safety and integration related issues.

Design Team Interaction

In order to facilitate interaction among students, several techniques were used. A list was maintained with telephone numbers and email addresses of the students involved with the project at the time so that members could contact each other as issues arose. In addition, a team meeting was held weekly. At this meeting, a status review was presented and then each subsystem lead gave a short summary of the activities that had been accomplished. Short questions were answered and then the meeting was adjourned to allow subsystem members to interact with the other team members.

To communicate with the rest of the ION-F group, several methods were used. Each week, a teleconference (telecon) call was held where the system engineers and PI's would be present. This was an effective way of relating new information that came down from program management and for raising concerns about the design process. Since the calls were held real-time, it was often possible to achieve a resolution in shorter time that was possible using email. In addition to the systems group, several subsystems also held telecons among smaller groups that allowed for discussion of specific problems.

A number of email list servers (listserv) were established by VT for subsystem design teams to use. Students were encouraged to subscribe to the listservs that applied to their designs. In this way, ideas could be rapidly spread and documents could be passed along for team members to read.

A File Transfer Protocol (FTP) server was maintained at USU and http web servers were maintained by VT and UW. These servers contained important program documents and working group level documents. These servers were accessed by members of the ION-F group so that they could obtain information about the latest status of the design. In addition, design group members were encouraged to upload any new results or simulations to the FTP server. Backups of the servers were completed periodically in order to preserve the information available.

One last method was used to promote team interaction. Technical Interchange Meetings (TIM) were held after major team milestones. To resolve issues that could not be efficiently resolved in another way, students and PI's would travel to one of the universities for a major meeting. These meetings would take place over a weekend at a convenient point during the semester.

Schedules and Documentation

This area was one that held the largest challenges for the USUSat design team. For any complex space system, a minimum level of documentation must be generated and for most programs, the documentation is extensive. Safety engineers must be able to review designs in order to determine that no uncontrolled hazards threaten the Shuttle. In addition, team members must be able to review the work done by others in order to ensure that the designs are compatible. Often documentation reviews can pinpoint areas where two or more groups are working on the same problem or where no groups are addressing an issue.

USUSat was designed to be a very low cost program and to use a large amount of student and volunteer involvement. Graduate students were chosen for team leads wherever possible to

ensure continuity, but a large number of volunteers and undergraduate students also were involved in key areas of the design. This became problematic when trying to obtain documentation since many people would not allocate time for this purpose. Students were often concerned with class activities, exams, current employment and finding permanent employment. Volunteers would often participate in design and testing activities, but writing documentation held very little appeal and was not appreciated as an important part of the design process; it was ignored more often than not.

As a result, system engineers or PI's would often have to sit down with the person and take notes while speaking about the design. These collected notes were sometimes all the documentation available about a design at certain points. Some students were also very good at completing documentation and would provide written updates once or twice a month about the status of their subsystem.

Generally, the best way of completing the design was to establish an interface document under the control of a key team member and to have this member track information such as mass, volume, electrical and mechanical interfaces with individual components. A "wiring bible" was completed at SDL by a technician and major team members. This document indicated the electrical connections maintained in the spacecraft. Mechanical interfaces were tracked in an I-DEAS software model and drawings were then generated and checked into the SDL documentation control system. Changing these interfaces then required the approval of a qualified engineer.

In addition, a large spreadsheet was maintained that tracked many of the major interfaces. It assigned components to be controlled by a specific person or entity. In addition this spreadsheet tracked expected materials usage, expected budget and an expected schedule. This spreadsheet was constructed by program management with inputs from the design team members. A sample of the spreadsheet is shown as Table 5 and the full spreadsheet is available in Appendix I.

In addition to this overall control document, USUSat engineers tried to maintain an overall schedule that they could use to track problem items. This schedule was maintained using Microsoft Project software. This schedule was built using inputs from this master interface control document and also from program level schedules. In this way, management could see how the status of their project was progressing with respect to the goals of the overall project. The schedule is shown in Figure 13.

Table 5. USUSat Master Interface Control Document

Component Name	Engineering Responsibility			Mechanical Specifications				Electrical Specifications						
	ICD Reference #	Mechanical Engineer	Electrical Engineer	Mass (g)	Size	Thermal Range Storage	Thermal Range Operational	Schematics (File Name)	Connectors and Pinout Drawings	On Power Draw (W)	Orbit Averaged Power (W)	Voltage	Duty Cycle	Other
Science Components														
PIP & DC Electronics board	2.1	NA ↑	Chad Fish	100	Card size (7)	-40 C to 85 C	-40C to 85C	PIP @ SDL	TBD		1.50	+/-5+/-12	100%	
Structural Components														
Top Panel	3.1	Bret Ashby	NA	29507	Major diam. 19.375", minor diam, 6" tall	NA	NA	SGI /nanosat/bja/newmodel.mf1	NA	NA	NA	NA	NA	NA
Bottom Panel	3.2	Bret Ashby	NA	included in above	Major diam. 19.375", minor diam, 6" tall	NA	NA	SGI /nanosat/bja/newmodel.mf1	NA	NA	NA	NA	NA	NA
Side Panel 1 (Nadir)	3.3	Bret Ashby	James Gutshall	included in above	9.875 " x 5.5" x 0.375 "	NA	NA	SGI /nanosat/bja/newmodel.mf1	NA	NA	NA	NA	NA	NA
Side Panel 2	3.4	Bret Ashby	NA	included in above	9.875 " x 5.5" x 0.375 "	NA	NA	SGI /nanosat/bja/newmodel.mf1	NA	NA	NA	NA	NA	NA
Side Panel 3	3.5	Bret Ashby	NA	included in above	9.875 " x 5.5" x 0.375 "	NA	NA	SGI /nanosat/bja/newmodel.mf1	NA	NA	NA	NA	NA	NA
Side Panel 4 (Zenith)	3.6	Bret Ashby	NA	included in above	9.875 " x 5.5" x 0.375 "	NA	NA	SGI /nanosat/bja/newmodel.mf1	NA	NA	NA	NA	NA	NA
Side Panel 5	3.7	Bret Ashby	NA	included in above	9.875 " x 5.5" x 0.375 "	NA	NA	SGI /nanosat/bja/newmodel.mf1	NA	NA	NA	NA	NA	NA
Side Panel 6	3.8	Bret Ashby	NA	included in above	9.875 " x 5.5" x 0.375 "	NA	NA	SGI /nanosat/bja/newmodel.mf1	NA	NA	NA	NA	NA	NA
Honeycomb panel		Bret Ashby	NA	TBD	TBD	NA	NA	SGI /nanosat/bja/newmodel.mf1	NA	NA	NA	NA	NA	NA
Fasteners and helicoil		Bret Ashby	NA	TBD	TBD	NA	NA	SGI /nanosat/bja/newmodel.mf1	NA	NA	NA	NA	NA	NA
Stack eye bolts		Bret Ashby	NA	TBD	TBD	NA	NA	SGI /nanosat/bja/newmodel.mf1	NA	NA	NA	NA	NA	NA

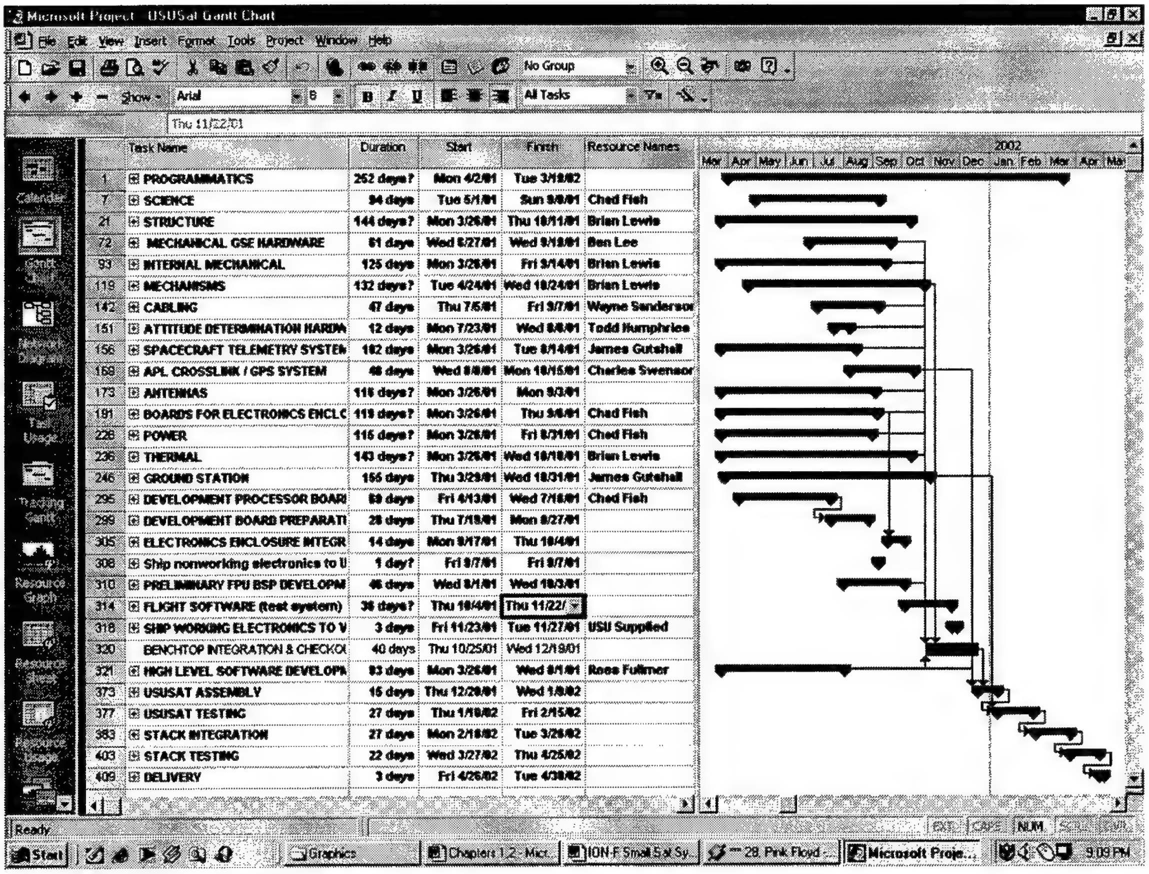


Figure 13. USUSat gantt chart.

This chart does not show the full extent of the schedule that management had drawn up. The entire gantt chart is included in Appendix 1. Management attempted to create a thorough, realistic schedule that would take USUSat to completion. Each aspect of design and test that management and subsystem engineers anticipated was included; and time estimates were made to reflect the reality of working with a student project.

While some mistakes were made in the management aspects of USUSat, it was not due to a lack of effort or desire. Rather it tended to reflect the fact that most engineers are taught to design systems of hardware rather than to manage projects. Being a student project, most work was done irregularly and in spurts. During exams or times when major class projects were due, work on USUSat was almost nonexistent. In addition, most students were not paid to work on the project and as such had other time commitments to their current employers.

Students, this author included, also tended to overestimate their abilities or underestimate the scope of the work that they were trying to complete. As such, the schedules that many would agree to meet were ultimately unattainable. Designs were also often taken from textbooks or from other experience and students had little experience in making these book designs a reality. In retrospect, having a manager dedicated to the project or using students studying business management to serve as project managers would have been helpful in completing the project on time. A clear definition of student responsibilities and realistic, workable job assignments would have helped many students to complete their designs and avoid being overwhelmed (Hansen, Summers, and Clapp 1991).

CHAPTER 3: SPACECRAFT SUBSYSTEMS AND DESIGN CHARACTERISTICS

USUSat Structures

The USUSat structural design was described in detail in the thesis written by Bret Ashby (2001). Therefore, this explanation will not go into great depth as his thesis is available to describe the design. As stated previously, the design of the structural subsystem was meant to be simple. The design originally had two deckplates with six corner struts and six side panels to be made from 6061-T6 aluminum sheet. The internal components would be mounted either to the lower deckplate as shown in Figure 4 or to the side panels as shown in Figure 14 below. Three eyelets were included on the top panel in order to provide capabilities for lifting the ION-F stack as requested by AFRL.

However, while performing subsequent calculations, the structural design team found that simply using aluminum plate and sheet exceeded their allowable mass budget. By using thin enough sheet to meet their mass budget, the strength and stiffness of the structure dropped below acceptable margins.

Three approaches were considered as solutions to this problem. First, engineers could choose to design an isogrid structure. Isogrid is a design in which a pattern of triangular pockets are cut out of a solid plate. In USUSat's case, the pockets were not cut completely through the plates, but stopped short leaving an external skin. By cutting these pockets out of the plate, a network of support ribs, similar to a truss structure, was left that provided stiffness and strength comparable to thick plate while being reduced considerably in mass. This had the unfortunate side effect of being considerably more difficult to design and machine. In addition, mounting locations had to be carefully placed rather than being located as desired.

The second option was to produce an isogrid structure but instead of leaving the skin in the piece, engineers would cut completely through the plate to form a truss structure. A thin sheet of aluminum could then be epoxied to the outer surface. In this way, thinner skin could be designed than if traditional machining methods were used. This would reduce machining time and mass. Finally, the teams could identify an aluminum honeycomb and produce the structure from honeycomb panels.

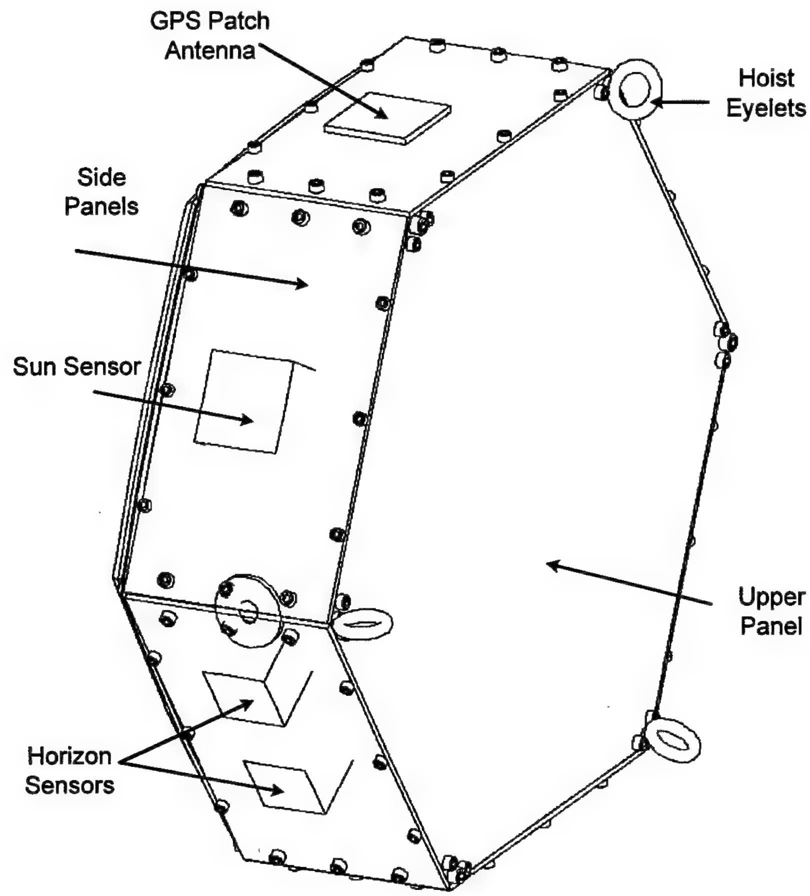


Figure 14. Preliminary external design.

Each approach had several advantages and disadvantages associated with their use. The traditional isogrid panels would be time consuming to design and manufacture but were a technology that was familiar to NASA safety engineers. The isogrid using epoxied sheets would be faster to design and fabricate, but would have to be classified as composite structural elements under NASA safety directions. This would require extensive testing and carefully supervised assembly methods. The honeycomb structure would be lightweight and would not require intensive design efforts. However, special inserts would have to be obtained for placing fasteners. In addition, the honeycomb would have to be obtained from a manufacturer that was approved by NASA engineers; procurement would increase costs significantly. Since USUSat could come very close to meeting its mass budget with traditional isogrid, this option was selected for the final design. UW and VT engineers selected the epoxied isogrid for their spacecraft.

Figure 15 shows one of USUSat's six side panels with the isogrid pattern clearly shown. This panel also shows a cutout reserved for the deployment of one of USUSat's two booms. Notice the extra machining detail required for mounting positions.

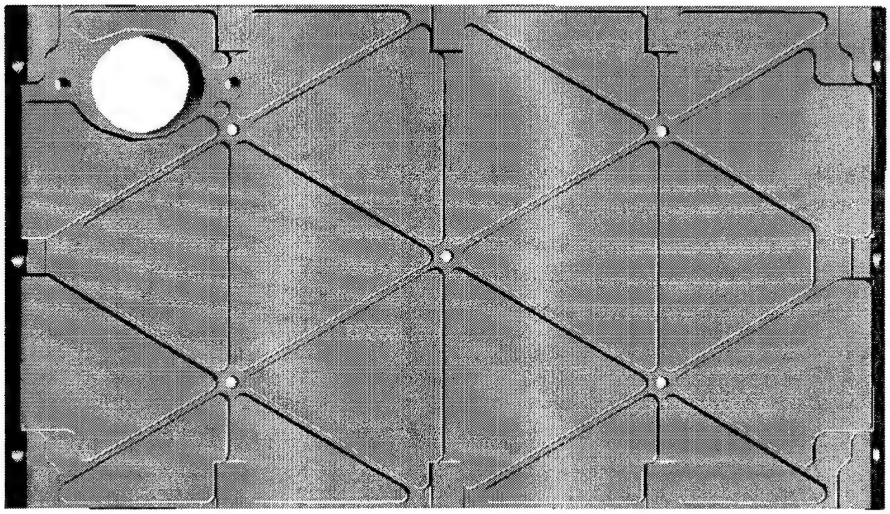


Figure 15. Isogrid pattern on USUSat side panel #6.

All of USUSat's panels were similarly redesigned using 6061-T6 aluminum with threaded helicoil inserts for fasteners. One additional change was made to the lifting hardware used. Personnel from STP recommended that swivel rings be used instead of eyebolts as the swivel rings would reduce lateral loading in the structure and help make lifting procedures simpler for ground support personnel at AFRL and USU where the lifting would occur.

In order to verify the structure's ability to withstand launch imposed loads and vibration, two methods were used. One, a finite element model was constructed using the I-DEAS software package. Two, a prototype structure was machined that could be dynamically tested using a shaker table available at Space Dynamics Laboratory (SDL) near the USU campus. This prototype structure is shown in Figure 16.

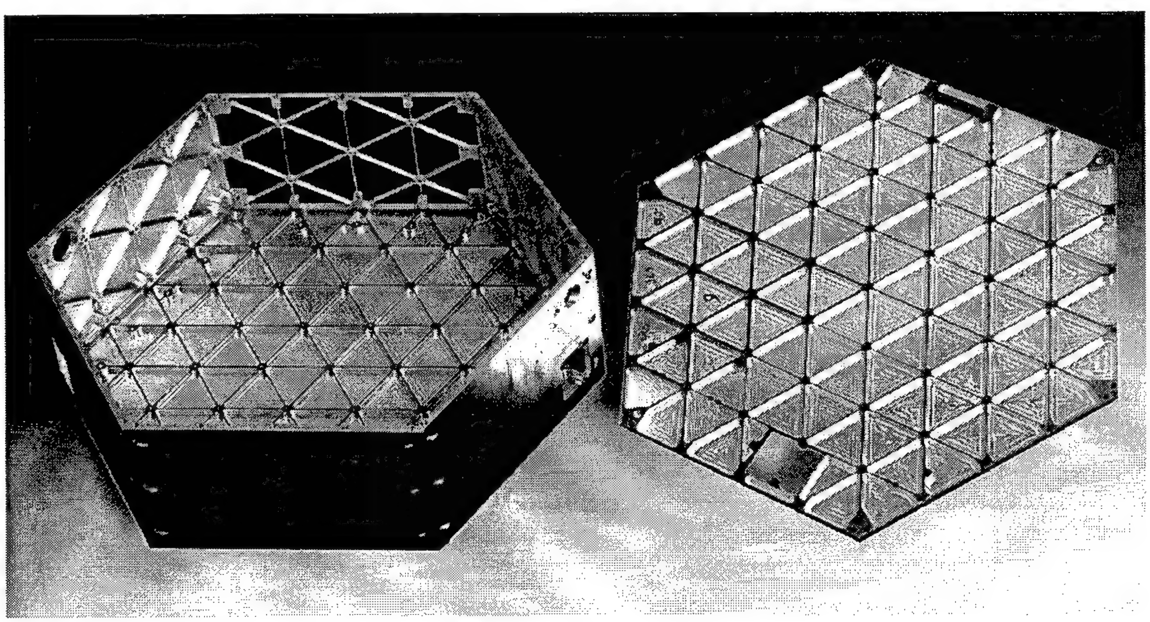


Figure 16. USUSat prototype structure.

The finite element analysis and vibration data predicted similar results except in one area. The finite element model predicted that safety margins would be inadequate and that the bottom structural panel would experience yielding under maximum loads. When the actual structure was subjected to a vibration test, no yielding or deformation was found in the areas predicted by the model. Still, systems engineers decided to use 7075 aluminum instead of the 6061 aluminum previously specified. The 7075 aluminum has higher strength, but is more expensive. By machining the one panel of concern out of 7075 and leaving the rest 6061, program costs could still be kept relatively low while ensuring that safety margins would be maintained.

One last change was made to the structure after this point before final machining of flight parts was carried out. Thermal engineers that were using the prototype structure in separate tests discovered that there was insufficient heat transfer out of the computer and electronics case and that electronic parts were exceeding their operating temperatures. Engineers determined that if the bottom panel could be redesigned so that the computer case was integral to the bottom structural panel, the heat transfer would be sufficient to keep the electronics cool. The bottom panel was redesigned to accommodate this request. During the redesign two issues were raised. One, the Space Dynamics Lab machine shop was uncomfortable with machining 7075 aluminum since stock was not readily available and new stock would have to be specially ordered. In response to this, engineers determined that the extra material placed into the bottom panel as a result of the redesign should have significantly stiffened the panel. As a result of these issues, it was decided to machine the bottom panel out of 6061 aluminum as previously specified.

USUSat Structures Subsystem Budget

A list of the parts used in the structural subsystem and their estimated masses is shown in Table 8. It can be seen that actual part masses greatly exceeded their initial budgets. As described above, preliminary designs utilized aluminum sheet with corner struts. When it became apparent that these were inadequate, the budget was reevaluated. The structures design team was given a new total mass budget of 3.4 kg for the structural subsystem, but even this budget has been exceeded. To explain this discrepancy, we note the difference in mass as given to the nadir panel and Lightband. Due to an error that arose in the interpretation of PSC design documents, USUSat was originally designed for a Lightband half that had around 680 g of mass while Dawgstar was to receive the heavier half with a mass of 811 g. After Dawgstar had been machined, fit checks revealed the error. As a result, Dawgstar had to use the lighter half while USUSat used the heavier. In addition, the bottom panel had to be redesigned to integrate the flight computer enclosure and to provide additional stiffness.

Power subsystem engineers also calculated that large portions of the solar cells on the bottom panel would be shadowed by the Lightband system. In order to prevent shadowing, honeycomb extenders were found that would raise the cells off the panel surface. Finally, some mounting parts that were unanticipated early in the design had to be reincorporated later. It may be tempting to assume that structural engineers could have optimized the isogrid patterns further in order to reduce mass, but manufacturing techniques and facilities available dictated the minimum size of the isogrid parameters. USUSat engineers felt that even though the budget had been slightly exceeded, the effort required to further reduce the mass would have caused

unacceptable budget overruns, time delays, and increased safety monitoring and paperwork. The structural subsystem was deemed acceptable and it was manufactured.

Table 8. Structural Subsystem Mass Budget

Subsystem	Component	Predicted Mass (g)	Actual Mass (g)
Structures	Base Plate	454.0	725.0
	Top Plate	454.0	625.0
	Side Panels	454.0	1344.0
	Lightband	680.0	811.0
	Fasteners	181.0	580.0
	Total	**2223.0**	**4085.0**

USUSat Mechanism Design

The design of USUSat incorporated two moving mechanical systems. One was a set of deployable booms, to be used for science experiments and in attitude determination; and the second was an actuated magnetic gimbal that would be used for attitude control. A third system, a set of deployable antennas, was included in the design for some time but was removed for reasons that will be described below.

Deployable Booms

The deployable booms were required on USUSat for two reasons. The first had to do with the spacecraft's science mission. Measurements were to be taken of plasma and ion densities and frequencies in the upper atmosphere, called the ionosphere. This plasma heavily affects the behavior of radio waves. Better understanding of how this plasma behaves can help engineers design better communications systems in the future. While a spacecraft is traveling through this plasma, it absorbs some of the electrons that make up the plasma, causing the spacecraft to become negatively charged. The spacecraft therefore produces a wake and a bow wave, similar to that produced by a boat in water, as it moves through the plasma. For these reasons, accurate plasma measurements must be made some distance away from the spacecraft's surface and in the ram direction so that they will not be contaminated by the spacecraft itself. For these reasons, a deployable boom will be used on USUSat in order to obtain accurate science data.

The second reason that booms are needed is for the attitude determination system (ADS). The ADS system uses a three axis fluxgate magnetometer to determine the magnitude and direction of the earth's magnetic field. However, the attitude control system (ACS) relies on large permanent magnets for control actuation. These magnets corrupt the field strength readings obtained by the magnetometer and yield false information about spacecraft position and attitude. In order to obtain correct readings, the magnetometer must be moved some distance away from the control magnets. Thus, the magnetometer will be placed into the tip of one deployable boom.

As shown in Figure 4 previously, the deployable booms were originally conceived as being spring loaded booms that would be mounted across the midsection of the spacecraft. The springs would propel the booms out approximately 20 inches from the edges of the spacecraft. Since this is longer than the full diameter of the spacecraft, the booms would have to be segmented, similar to telescoping antennas used on automobiles. During an early program status

review, NASA engineers asked nanosatellite designers to remove stored energy sources wherever possible, so a new method of deployment was needed.

Designers then turned to a gear driven mechanism. A small stepper motor would be used to drive a worm gear attached to two pinions. The two pinions would drive two long lead screws. The boom would interface with the lead screws so that when the screws were rotated, the boom would be driven out of the spacecraft. Two changes were also made to the booms in addition to the deployment method. Due to the size of the flight computer enclosure and some of the crosslink communications gear, the booms could not occupy the central area of the spacecraft. They were redesigned to be small enough to fit over the top of the computer and communications gear. Also, the requirement for boom deployment length was relaxed to around 15 inches and the booms were redesigned to use single pieces rather than multiple segments. The gear system used to deploy the booms is shown in Figure 17. In this figure one lead screw and the main boom itself are hidden for clarity.

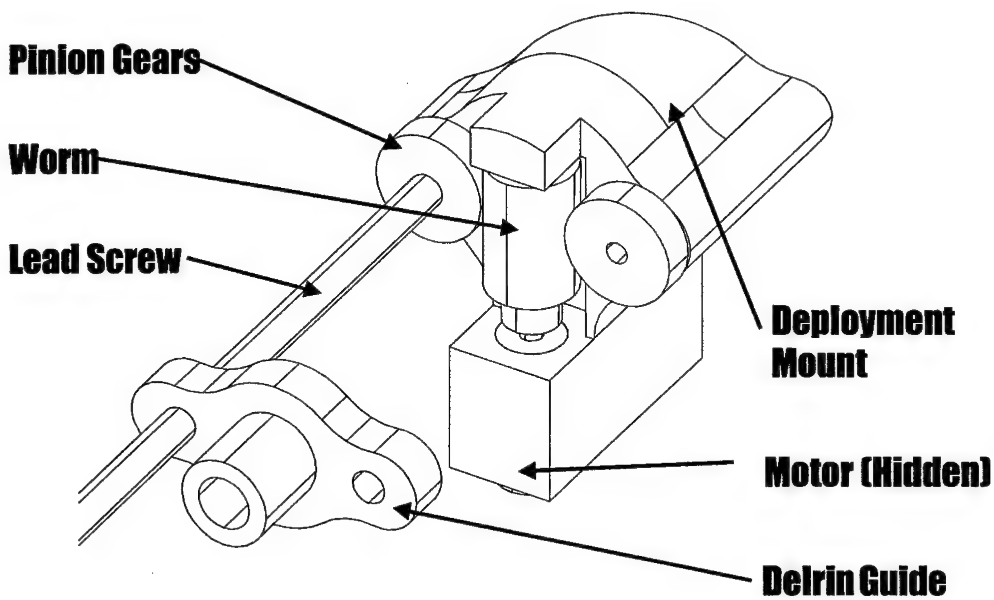


Figure 17. Boom deployment gear system.

In this configuration, designers were relying on two design features in order to prevent inadvertent deployment. The first was through the use of the worm gears, which cannot be back driven when the worm lead angle is less than a critical angle. The second was the fact that the stepper motor itself had an internal detent torque. This torque served as an initial threshold. Externally applied torques had to be larger than the detent torque in order to cause the motor to rotate. In each case, designers took the proper steps for success. Gears were selected with lead angles sufficient to prevent back drive and all externally applied torques due to the launch environment were less than the detent torque with a sufficient safety margin.

Unfortunately, during a program review, NASA engineers decided that both of these types of boom retention could be classified as friction brakes, which cannot be used to inhibit catastrophic hazards, a classification that will be covered later in this thesis. NASA engineers

stipulated that some sort of active retention using metallic parts must be present during launch. The retention mechanism must be removed from the path of travel after deployment from the Shuttle. The design team then worked to find some way of providing this retention. Several devices were considered before one was selected. The team decided to use an actuator called the Frangibolt manufactured by Tini Aerospace.

The Frangibolt device, shown in Figure 18, utilizes a shape memory alloy (SMA) cylinder interacting with a titanium bolt to provide the retention and release ability required. The bolt itself was notched with a particular profile. The actuator was slid over the bolt into the desired configuration. When the boom was to be deployed, an electric signal was sent to the device. The device heated the SMA cylinder causing it to expand. When it expanded, it stretched the bolt as well. When the bolt was sufficiently stressed, it would fracture across the plane defined by the notch. The notch served to weaken the bolt to ensure that it would break in the proper position and before any surrounding parts were broken as well. Proper design steps had to be taken in order to ensure that surrounding parts were sufficiently strong to withstand the forces applied by the actuator. The actuator was also selected so that its activation temperature was at least 10 °C greater than the maximum expected temperature while on the Shuttle.

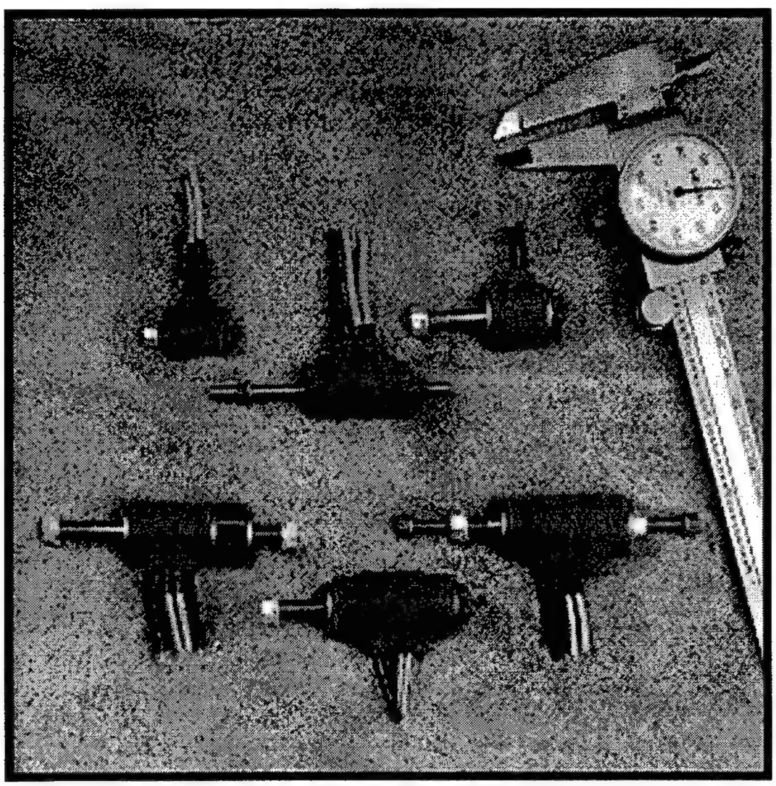


Figure 18. Frangibolt actuators.

These actuators were selected for a variety of reasons. Similar systems from other manufacturers were still being designed and would not be available in the required timeframe. The actuators were very sturdy and the smallest actuator, which was used in USUSat's design, provided very large safety margins. The actuators were small enough that only minimal redesign was required. The actuators had been used in spacecraft previously with success, which helped

alleviate NASA safety engineer's concerns. Finally, Tini Aerospace engineers were extremely helpful and offered some extra services to help student-designed projects. The booms were redesigned to incorporate the Frangibolt actuators. The redesigned boom deployment mechanism is shown in Figure 19.

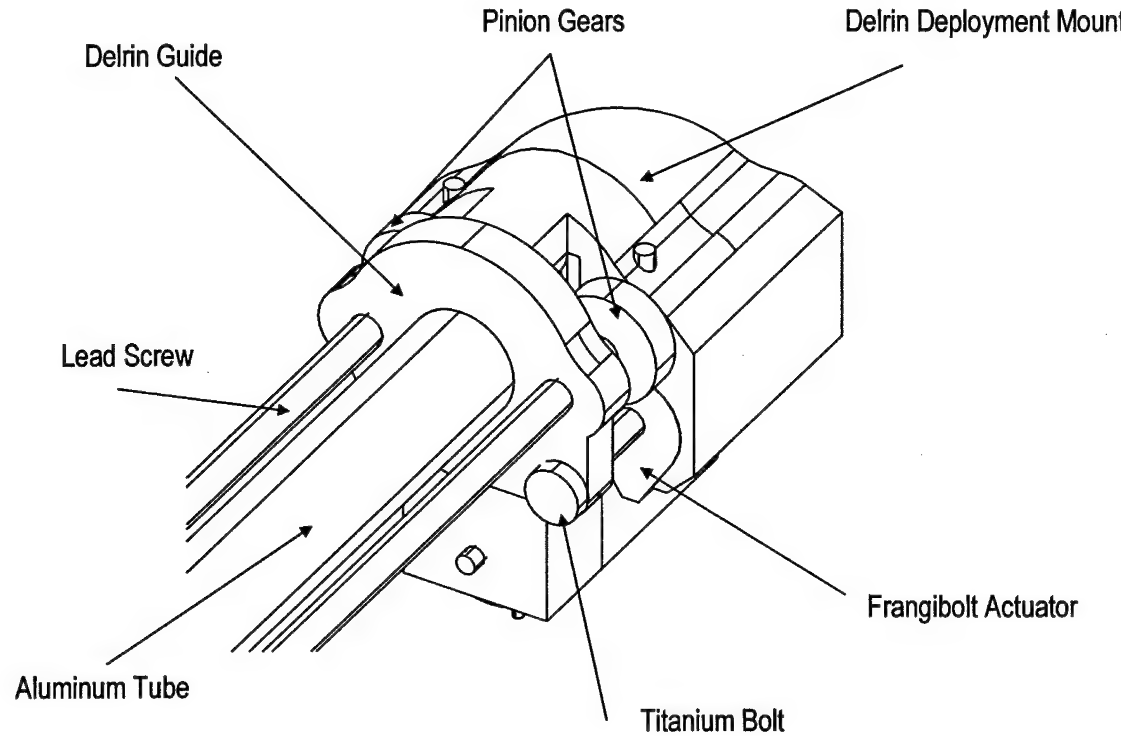


Figure 19. Boom deployment system with Frangibolt actuator.

As shown in Figure 19, the actuator was placed next to the stepper motor beneath the worm gears. The actuator would be fixed into place using set screws. When deployment was commanded, the actuator would break the titanium bolt and the gears would once again drive the boom out of the spacecraft.

One last problem confronted boom designers. Worm gear driven systems are very sensitive to alignment problems. USUSat designers were having problems properly aligning the gears correctly. The misalignment problems were causing higher than expected resistance torques and were preventing proper deployment. With the new Frangibolt parts incorporated into the design, USUSat engineers resurrected the idea of using spring loaded booms. Since the energy stored in the springs was relatively small and the Frangibolt actuators provided very large safety margins, this idea was tentatively accepted by AFRL program managers. Full approval must come from a NASA Payload Safety Review Panel (PSRP).

Magnetic Gimbal

As described above, the second mechanical system incorporated into USUSat is a control gimbal that relies on permanent magnets. The goal of experimenting with rotating permanent magnets was to determine if significant power savings could be realized when compared with the use of torque coils. This power savings would come at the cost of a system with multiple

moving parts that requires extra design and manufacturing effort. Figure 20 contains a conceptual idea of the gimbal design.

The gimbal was designed with three stepper motors. One motor would be attached to each magnet so that the magnets could be independently rotated through 360 degrees of motion. The motors and magnets would be mounted to a central arm that would be actuated with the third stepper motor. This would allow the magnetic vector of the gimbal to be pointed in any desired direction so that it could interact with the earth's magnetic field to rotate the spacecraft into the desired orientation.

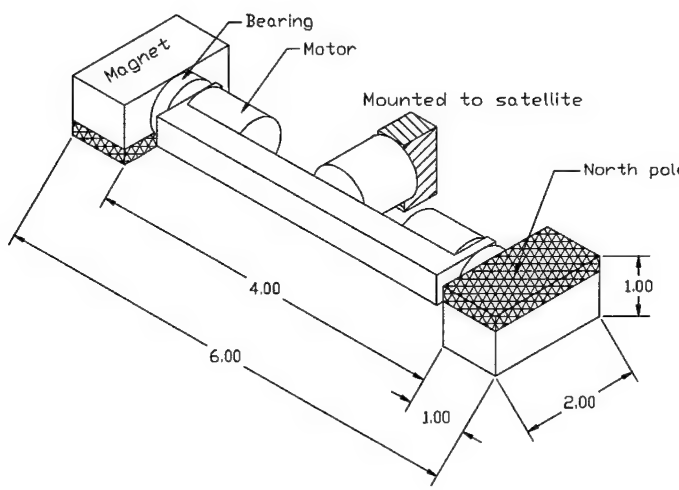


Figure 20. Conceptual drawing of magnetic gimbal.

One major problem exists in this design. Each of the motors has wiring connected to them. The two motors on the central shaft would have to have wiring that would rotate with the arm. Providing this wiring would present a challenge since it restricted the range of motion for the gimbal. The wiring was also prone to tangling and binding. Designers began to examine ways to build the gimbal so that the motors could remain stationary, thus eliminating these problems.

The solution to this problem came through the use of worm gears embedded within the structure of the gimbal itself. The motors would now be mounted on spindles attached to a central shaft. Each spindle would contain one worm pinion gear. Two worm gears ran through the center of the central shaft. The motors that drove these gears were now attached to the support structure that fastened the gimbal to the spacecraft structure. The central shaft itself could still rotate and another worm pinion was located on the end of the shaft. The shaft itself was capable of rotating when driven by a third worm gear that was also attached to the support structure. The use of these gears allowed the motors to remain in a fixed position.

One further addition was necessary. During some periods of the orbit, control engineers could not guarantee full three axis control due to the shape of the earth's magnetic field. In response, a small reaction wheel was included in the gimbal structure that would allow full control at all times. This redesigned gimbal is shown in Figure 21.

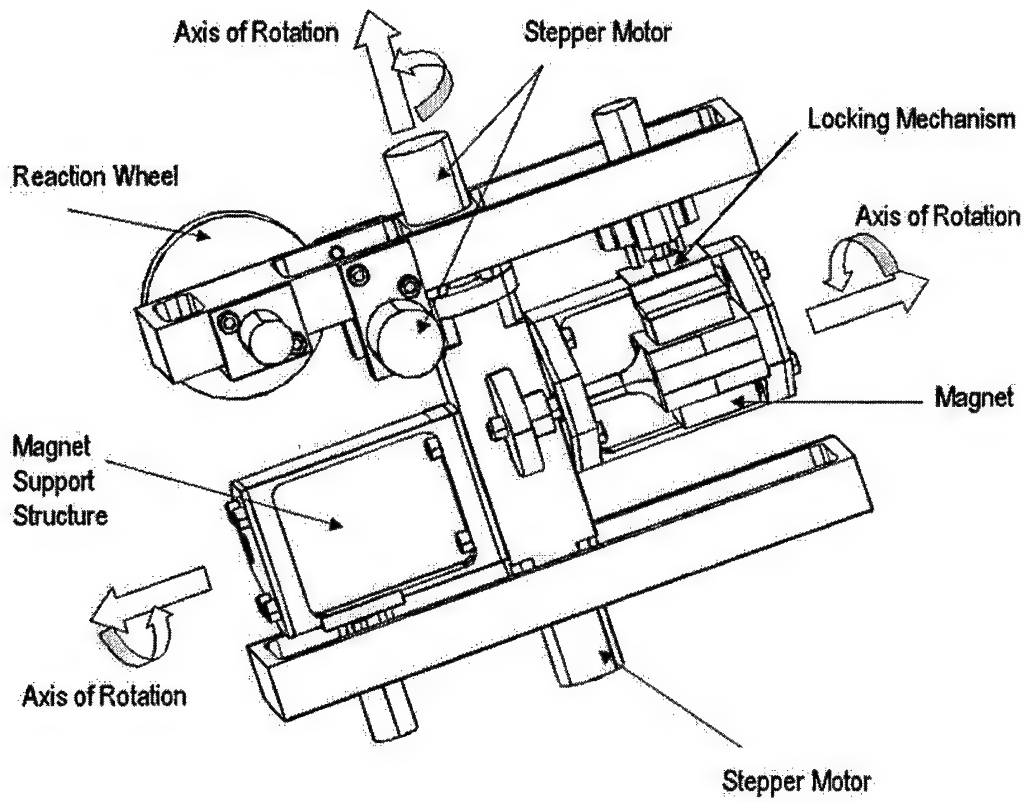


Figure 21. Gimbal with fixed motors.

This design was prototyped and some small errors were found. Some problems also existed in the tolerancing of gimbal parts that led to misalignment in the gearing. In addition, most of the shafts required in the gimbal were small and made of aluminum to avoid interactions with the magnets. Some of these shafts were inadvertently bent during assembly. Finally, certain pieces had been put into place using epoxy. There was no easy way to disassemble and reassemble the gimbal if required during testing. NASA engineers had requested the addition of locking mechanisms that would prevent the gimbal from unpowered rotation. Similar to the booms, they desired metallic locks that actively engaged the gimbal and that would have to be physically removed in order to rotate the gimbal. These locks are present in the design in Figure 21. However STP engineers asked for the inclusion of sensors that would indicate whether the locks were engaged or disengaged.

USUSat engineers returned to work to correct these problems and add the desired features. This time the machining would be done on high precision equipment by well trained machinists at SDL where the machining had previously been done by the USU Mechanical Engineering Department and mechanical engineering students in an attempt to save money. Shafts that had shown susceptibility were rebuilt from non-magnetic stainless steel instead of aluminum to increase strength. Cover plates that utilized threaded fasteners rather than epoxy were specified. Finally, contact sensors were placed in the "home" positions of the gimbal. The magnets must be in these positions for the locks to engage. To test for proper engagement, a command could be issued to rotate the magnets and the sensors could be queried. If the magnets remained in place, the locks had worked properly; otherwise, the locks had disengaged and would need to be repositioned. New drawings were made and the gimbal remachined.

Deployable Antennas

Earlier, it was stated that USUSat had only two moving mechanical assemblies. In early design phases, however, USUSat had a third mechanism: deployable antennas. Communications engineers had brought up the idea of using an emergency downlink beacon on USUSat. This beacon would transmit in amateur radio frequencies and would broadcast its signal worldwide. The signal would be rather simple, broadcasting basic spacecraft position and health data. Unfortunately, at the necessary frequency, the antennas required were too large to be simply fixed to the spacecraft. They would have to be deployable. Since they were proposing one set of deployables, the communications design team also proposed redesigning USUSat's uplink antennas as deployables as well. This would allow the spacecraft to receive commands from the ground in any orientation. The communications team wanted to use copper beryllium tape similar to a steel measuring tape that would unfurl into large deployed antennas nearly 40" across for the beacon and around 10" for the uplink antennas. These antennas would be placed on the nadir pointing face of the spacecraft. USUSat with the proposed deployed antennas is shown in Figure 22.

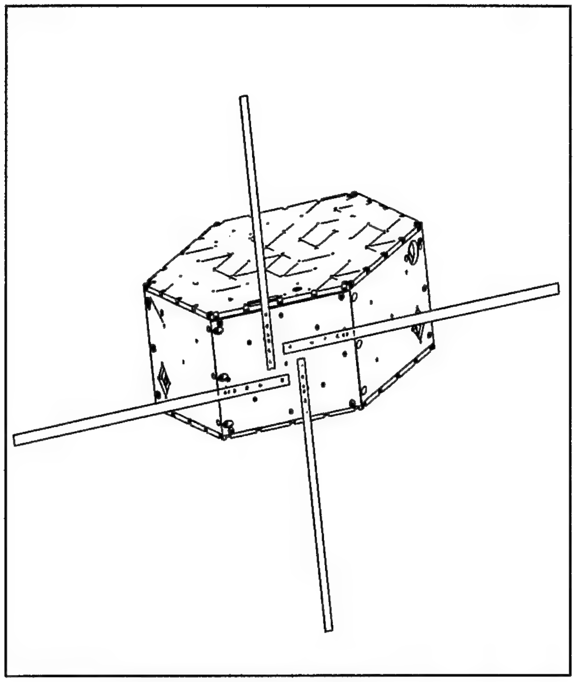


Figure 22. USUSat with deployed antennas.

The antennas would be coiled onto one of USUSat's exterior panels during launch and would then be deployed after the ION-F stack had deployed from the shuttle. The coils would be retained using nylon monofilament. The nylon was looped through small holes drilled into the copper tape. The nylon was then passed through to the back of the panel where it was brought into contact with a strand of nichrome wire. The two ends of the filament would then be crimped into place using a small copper crimp, similar to those used in electrical wiring. When the antennas were to be deployed, electrical current would be passed through the nichrome wire, which would heat up and burn through the nylon allowing the tapes to deploy. In order to isolate the antennas from the spacecraft to properly receive signals, an outer panel would be machined from Delrin. The aluminum isogrid underneath would have been machined completely through in order to save weight and a similar isogrid pattern would have been cut into the Delrin panel. This panel would have been attached to the aluminum beneath it using threaded fasteners. Figure 23 shows what the panel exterior would have looked like prior to antenna deployment. Figure 24 shows what the nichrome and crimp arrangement would have looked like on the reverse side of the panel.

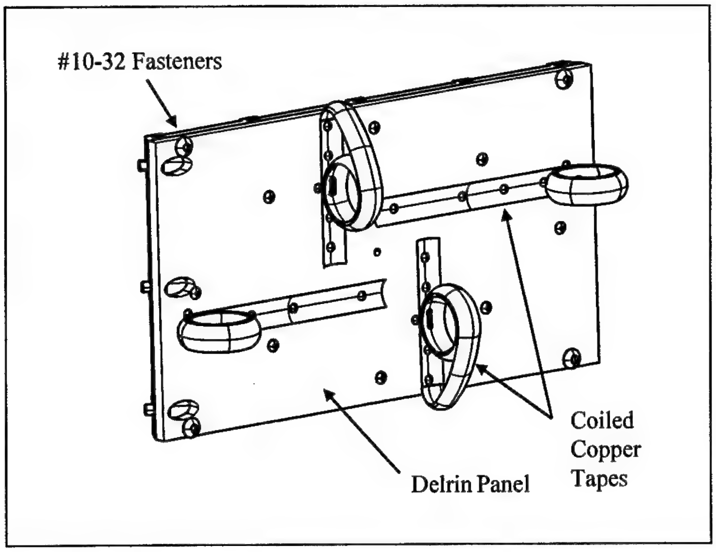


Figure 23. Deployable antennas in coiled configuration.

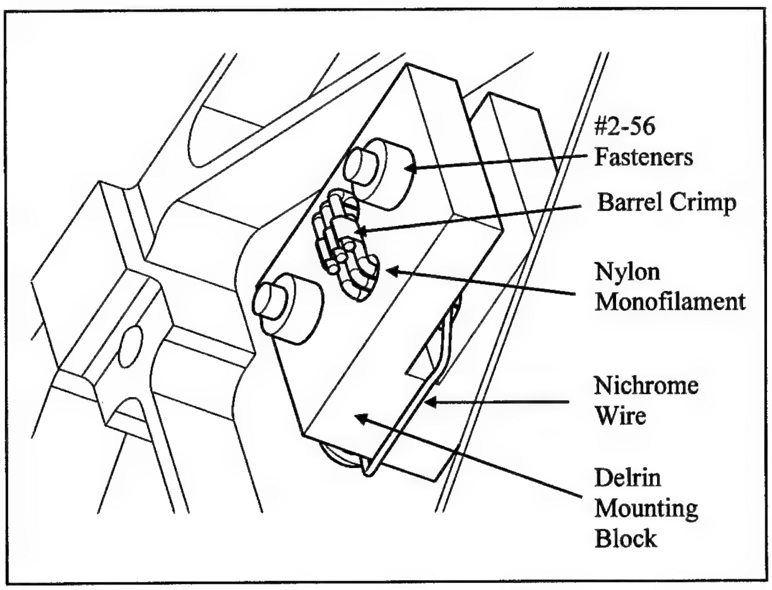


Figure 24. Nichrome mounting block.

NASA safety engineers had several objections to this design. The presence of nylon was unacceptable because it was a substance that was not extensively documented like most metallic materials are. There was uncertainty about the reliability of the line's strength in the stowed position. In addition, the manufacturing process was susceptible to workmanship in the method of assembly. The crimping process could have potentially damaged the nylon monofilament thus compromising its strength to an unknown degree. In addition, passing the filament through the structure and through the antenna segments exposed it to sharp edges that could damage the line thus compromising strength.

USUSat engineers went back to the drawing board to find a design that would be more acceptable. In the new design, the four sections of antenna were bent over once to a central point. At that point, a small post was placed onto the Delrin panel with a notch cut in the top. Holes were drilled into the copper tape and the holes were placed over the top of the post. A multifilament thread of a material called Vectran would then be strung through the notch in the post holding the tapes in place. The Vectran cord was then run through two holes in the structure. At one end, a pin puller manufactured by Starsys was mounted into a small bracket. A loop was made in the Vectran and this loop was placed over the pin of a pin puller. Near the other hole, a tensioner device built by PSC would accept the Vectran and be used to place the cord into tension. When deployment was to be initiated, the pin puller would activate and retract the pin. This would allow the Vectran cord to slip off the pin. The tension in the cord would cause it to contract toward the tensioner and slip off the outer post, thus letting the antenna deploy. The antennas in their looped configuration are shown in Figure 25.

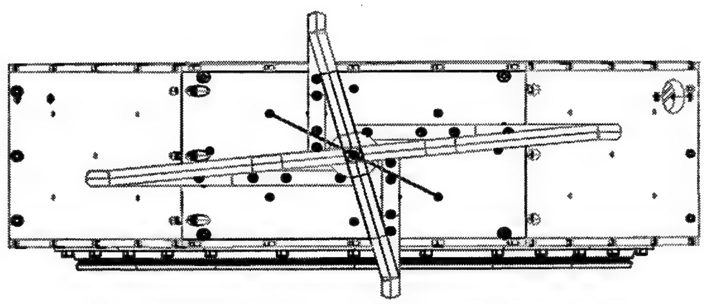


Figure 25. Looped antenna concept in stowed configuration.

Unfortunately, this design still posed problems. The Vectran cord was selected since it had been accepted in other space applications. These applications had conducted extensive testing of the cord during vibration and thermo-vac tests in order to determine how it was loaded, how it distributed the load, and what its failure modes were. NASA safety engineers wanted to see similar data for USUSat's application. In addition, NASA engineers were concerned about astronauts during extra vehicular activity (EVA) and for workers during integration. They felt that it would be easy for an astronaut or worker's tool to become entangled in the loops, thereby damaging the tool, suit, or antennas. They were also worried that the antennas could violate the spacecraft's dynamic envelope within the shuttle. At this time, communications engineers found a different solution that provided similar performance but that did not require deployables. This configuration is detailed in the communications section. When this design became available, the deployable antennas were completely removed from USUSat.

USUSat Mechanisms Subsystem Budgets

As can be seen, several iterations were required in order to produce designs that satisfied both program requirements and safety requirements. A list of parts and their masses compared with the budgeted masses is shown in Table 9.

The booms are slightly over their budgeted mass and this extra mass is attributed to the Frangibolt hardware required by NASA safety. The gimbal, however, has a much lower mass than originally thought. During the design of the gimbal, ACS engineers realized that the magnets they had originally selected were much larger than needed. They reduced the size of the magnets and the support hardware associated with them.

Table 9. Mechanical Subsystem Mass Budget

Subsystem	Component	Predicted Mass (g)	Actual Mass (g)
Mechanisms	Magnets	985.0	152.0
	Stepper Motors	181.0	318.0
	Gimbal Structure	680.0	370.0
	Electron Probe Boom	227.0	338.0
	Magnetometer Boom	272.0	358.0
	Deployment Actuators	91.0	87.0
	Total	**2436.0**	**1623.0**

Spacecraft Power

USUSat Power Systems

The power subsystem of USUSat was designed to help compensate for its mission profile. There is the possibility that USUSat will not be able to align itself for full charging during some of its orbits due to the formation flying mission. With this in mind, its power system was designed to produce extra power when possible.

Power Generation

USUSat uses body-mounted solar arrays for power generation. As discussed above, NASA engineers had requested early in the program that universities should use deployable systems only where absolutely necessary. On USUSat, five of its eight body panels are used for solar arrays. These panels should receive the most sunlight and are therefore the most efficient. The other three panels are nadir pointing and therefore do not normally receive much incoming solar energy.

USUSat uses Cascade Triple Junction developed by Tecstar. Many of the cells were purchased under an AFRL research program and so ION-F was able to procure the cells for a substantially reduced price. The cells are 23% efficient and the three layers are fabricated from $GaInP_2$ / GaAs / Ge. The cells are bonded to the body panels using a combination of epoxies from Nusil Technologies and are insulated with Kapton sheets. The overall assembly is shown in Figure 26.

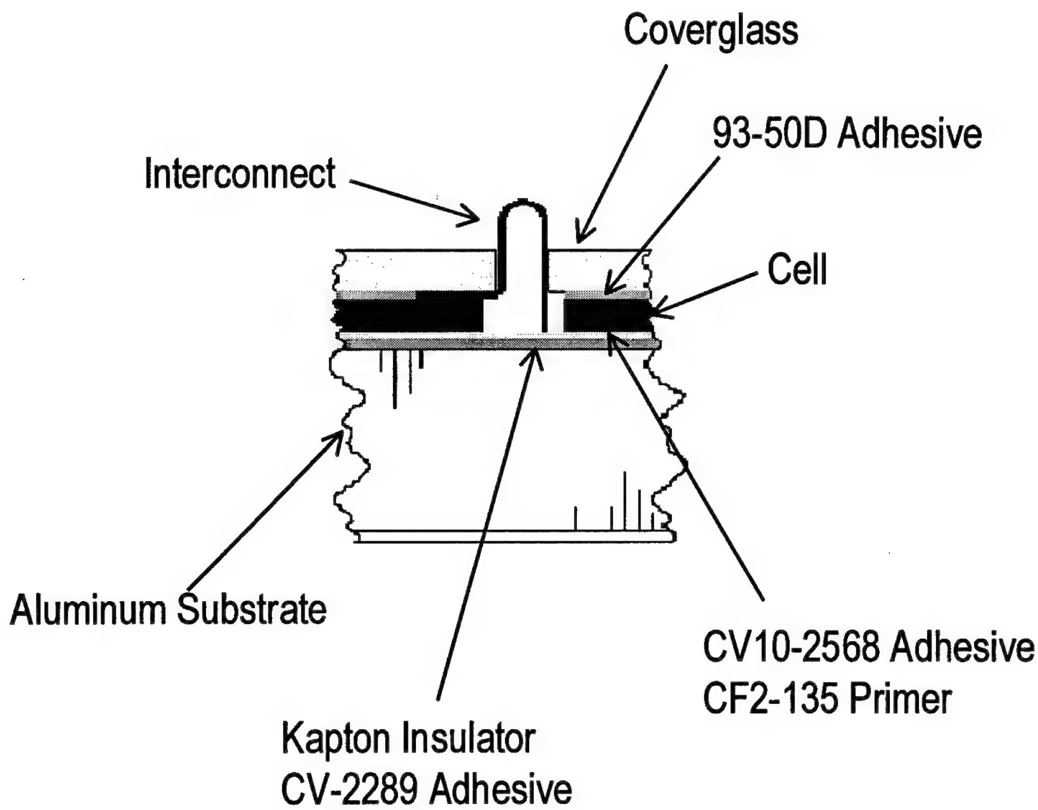


Figure 26. Solar array bonding.

For USUSat, the cells are joined together into strings of eight cells each. Ten strings of cells are mounted giving USUSat a total of 80 cells. 40 cells are mounted onto the large upper panel of the spacecraft. 16 cells are mounted on the bottom of the spacecraft in the center of the Lightband separation system. The three side panels that will face toward the sun most often have eight cells each. Each cell is 1.522" x 2.497" so USUSat will have around 304 in^2 of solar panels for energy collection. Due to half the cells being located on this face, the amount of power collected is highly dependent upon spacecraft orientation. USUSat will generate between 6 W and 18 W of total power with an average power generation of 9.5 W. This arrangement should give USUSat an unregulated bus voltage of around 18 V. The arrangement of solar cells on the structural panels of USUSat is shown in Figures 8 and 9.

The solar cell array fabrication took place at SDL under the direction of USU students Tyler Goudie and Ken Van Hille. In order to successfully fabricate the cell arrays, a multistep process was developed in conjunction with Tecstar, TRW, and SDL employees. The cells are first soldered into the configuration in which they will be installed onto the spacecraft. Kapton insulators are then prepared for the aluminum substrate that the cells will rest on. Adhesive is applied to the cells and to the substrate and the Kapton is bonded to the substrate and the cells applied to the Kapton. The sections of substrate are then vacuum bagged and cured for 18 – 24 hours. Figure 27 shows one section of solar array undergoing fabrication.

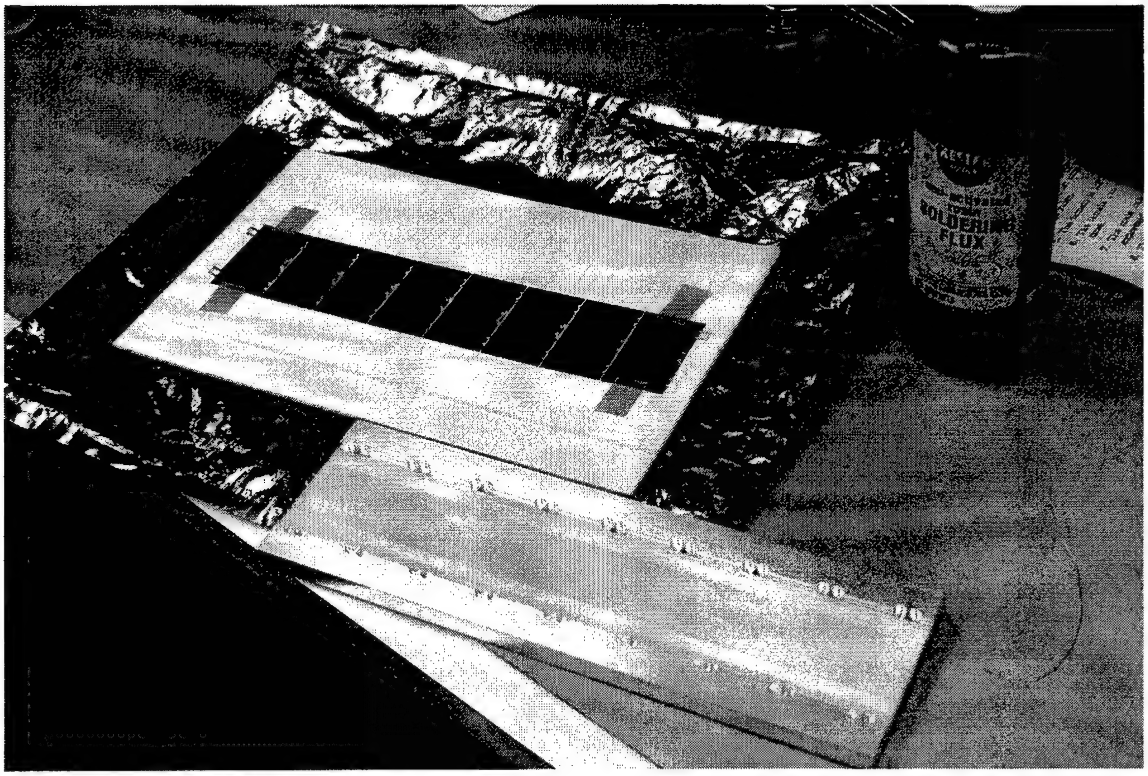


Figure 27. Solar array fabrication.

Power Storage

As mentioned before, the power system was designed to provide ample power even during periods when minimal charging was available. This meant that the USUSat battery would need sufficient capacity to allow for deep depth of discharge without showing significant degradation. USUSat chose to use NiMH batteries from Sanyo. The model selected was the Sanyo HR-4/3 FAU. This model was selected since it had been used previously on other Shuttle missions.

Eleven batteries would be strung together in series to attain the same unregulated bus voltage provided by the solar cells. The cells would be capable of providing approximately 4.5 Amp-hours of capacity. This was estimated to be a 300% margin over anticipated needs. The cells would be welded together into a pack and carried in a box designed by engineers at Virginia Tech. Battery and box design is extremely important to NASA safety engineers as it has been a common failure mode in the past. Virginia Tech was in charge of safety for the ION-F stack and so it was decided that the boxes and packs would be designed there so that safety concerns could be worked out rapidly.

The boxes were designed to be fabricated from 6061-T6 aluminum. They also were required to have coatings on the interior that were non-conductive and corrosion resistant so that electrolyte leaks due to faults in the electrical system would be contained. The boxes were designed to have an interior coating of nickel and solethane. This combination would meet the conduction and corrosion requirements. In addition, safety requirements mandated that there be a form of potting material around the batteries that would absorb any electrolyte leakage. The potting material that was selected is known as Pigmat and is essentially a polypropylene paper towel. The amount of electrolyte was compared to the absorptive capacity of the Pigmat and

sufficient Pigmat was included to absorb all electrolyte leakage. In addition, the boxes had to have vents to prevent the buildup of gases within the battery boxes. Often, when batteries fail, hydrogen gas can be generated and this gas must be vented before it can build up to flammable concentrations. The vents also had to be designed to preclude any liquid leakage while allowing for gas venting. The vents also had to be located above the centerline of the box when it was in its launch position within the shuttle. Finally at least two vents were required and they had to be sized so that either vent could vent any built up gas concentrations. The vents selected are from Osmonics and are built of Teflon with micropores that would allow gas to vent without allowing liquids to escape. A drawing of the battery box is shown in Figure 28.

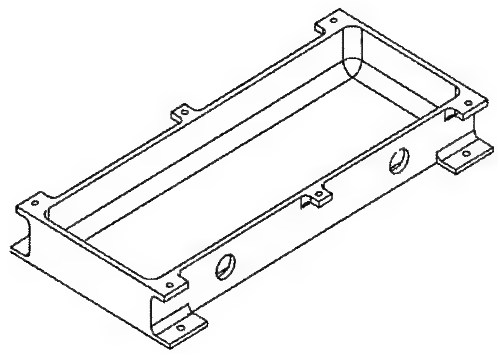


Figure 28. Battery box frame.

Power Distribution and Regulation

USUSat's power system was designed to provide unregulated voltage power from the solar cells and batteries and then convert the power into the forms that would be usable by the spacecraft subsystems. In addition, the system was responsible for preventing distribution of power before it was desired. An overall schematic of the power distribution within USUSat is shown in Figure 29.

The power system is responsible for providing 28 V, ±12 V, 10 V, ± 5 V, and 3.3 V power supplies as well analog and digital grounds for spacecraft subsystems. It has to be capable of providing large, one time power draws for deployable mechanisms as well as small, well regulated power for nearly continual applications.

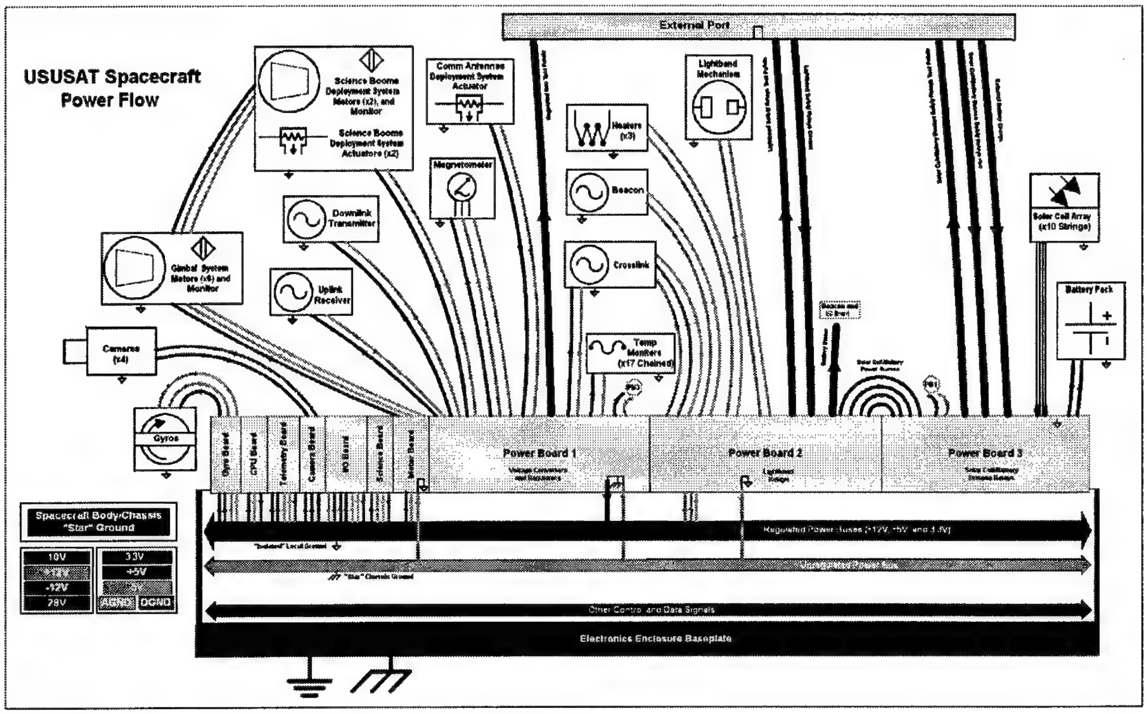


Figure 29. Power distribution within USUSat.

The power system has been designed as a DET system where any excess electrical power will be sent through the solar cells to be re-radiated to space. In addition, the power system has been designed with a series of relays that prevent undesired power flow from either the solar cells or the battery to the spacecraft bus. It can also prevent undesired flow to the deployable systems. This functionality was provided to meet the USUSat mission profile as discussed in Chapter 2.

USUSat Power Subsystem Budgets

The power system has been designed to meet the requirements placed on it by its mission goals and safety requirements. It is possible to compare the current mass budget and power budget with initial estimates to see how close the final design came to the expected system. The mass budget for the power subsystem is shown in Table 10 and the power budget is shown in Table 11.

Table 10. Power Subsystem Mass Budget

Subsystem	Component	Predicted Mass (g)	Actual Mass (g)
Power	Power Regulation	45.0	819.0
	Solar Cells	455.0	398.0
	Batteries	2725.0	1358.0
	Cabling	905.0	2750.0
	Total	**4130.0**	**5325.0**

Again, the actual design uses more mass than was initially budgeted. This is interesting since the batteries, box, and solar cells all were able to be designed smaller than initially

estimated. The big gain here is seen in the wiring category. These are both estimates since the software used could not predict the wiring harness mass accurately. Two SDL employees who had years of experience with wiring harnesses for both satellites and sounding rocket payloads made estimates in the range shown in Table 10. This estimate is conservative, and the actual harness should have less mass than what is predicted here. It is therefore probable that the power subsystem will actually have a mass rather close to that originally estimated.

Table 11. Peak and Average Power Budgets for USUSat

Component	Est. Peak Power (W)	Est. OAP Power (W)	Act. Peak Power (W)	Act. OAP Power (W)
Lightband	0.00	0.00	60.00	0.00
Stepper Motors	5.00	0.10	4.16	0.16
Deployment Actuators	0.00	0.00	28.00	0.00
Power Regulation	1.00	1.00	0.00	0.00
Kapton Strip Heaters	2.00	0.05	2.80	0.18
Temp. Monitors	0.10	0.01	0.00	0.00
S-Band Transmitter	8.00	0.05	28.00	0.34
Receiver	1.00	1.00	2.00	1.00
Beacon Transmitter	0.00	0.00	10.80	0.17
Data Formatter	0.10	0.10	0.88	0.38
Crosslink / GPS	2.50	2.50	10.20	7.18
Flight Computer	1.05	0.85	2.20	1.54
Data Buffer	0.23	0.03	0.32	0.18
CMOS Camera	1.50	0.50	1.20	1.00
Magnetometer	0.20	0.20	0.25	0.20
Sun Sensor	0.20	0.10	0.00	0.00
Control Electronics	0.40	0.10	0.91	0.41
Torquer Coils	6.00	0.05	0.00	0.00
Rate Sensors	2.00	1.00	0.00	0.00
Plasma Probe	1.50	1.50	2.10	1.50
Total	**32.78**	**9.14**	**153.80**	**14.24**

At first glance, it would seem that USUSat consumes too much power, especially in regard to what was originally budgeted. There are two major reasons for these discrepancies. First, the original budget did not include any power for boom deployment and for stack separation. Both of these activities require a large amount of power, but both will be performed only once during USUSat's mission. Therefore, these activities will drain USUSat's battery initially, but this power can be recharged once USUSat enters normal operation and there is lower, steady power consumption. The second reason for the average power consumption being rather high is in the design of the GPS receiver and crosslink system. This problem will be discussed later, but essentially, ION-F received extra funding to use specific crosslink hardware that another program wished to test before it was used on their spacecraft. This hardware was designed for larger spacecraft with larger power generation abilities. With these two discrepancies accounted for, the average power of USUSat actually comes close to its original budget. The 14 W of predicted power drain is larger than the average power generation that is expected. It is possible that due to this, USUSat will only be able to dedicate itself to formation

flying test for three or four orbits after which it will be forced to orient itself in a full sun-pointing mode that will allow it's batteries to be recharged. ION-F is also working with the designers of the crosslink hardware to try to reduce the average power consumption of this hardware.

Spacecraft Thermal Control

USUSat Thermal Control

The thermal analysis and design for USUSat was a complex problem and USUSat engineers turned to SDL for help. The analysis and design of USUSat's thermal subsystem was mainly performed by SDL engineers for this program and in conjunction with a similar project named Combat Sentinel. Combat Sentinel was a project in which military engineers wanted to test commercial parts for their hardness and survivability against hostile weapons attacks. A small craft similar to USUSat, but with around half the parts, would be placed into a thermo-vac chamber and shot with lasers while it was operating to determine how the components would respond to intense radiation. This project caused SDL engineers to take a keen interest in the thermal performance of USUSat (Moffitt and Batty 2002).

Thermal Analysis

The analysis of USUSat was conducted using the I-DEAS design software package. This package had also been used for the structural design of USUSat and as a result could be used easily by thermal engineers. A preliminary analysis conducted predicted that USUSat would see temperatures from around -29° C to around 20° C. Internal components that were the most vulnerable saw a much smaller temperature range, but in general the spacecraft seemed to be slightly cold biased.

Thermal Design

The analysis performed by SDL seemed to indicate that surface coatings and a few small Kapton strip heaters would be sufficient to maintain USUSat hardware within its operating temperature range. Engineers decided to use a paint that had been used previously at SDL called Aeroglaze A276. Minco heaters HK5411R236L12 were selected to provide around 1 W of heat in small areas. Finally, a series of small temperature sensors from Maxim semiconductor were selected to monitor thermal performance in USUSat.

When Combat Sentinel was put into use however, a problem was found in some of the computer system electronics. While most systems were performing as expected, some of the computer parts were overheating. The heat transfer from their boards through the box built to contain the computer and associated electronics was insufficient. In response, the computer box was redesigned to be an integral part of the spacecraft's bottom panel as described previously. Further modeling has predicted that this redesign will provide sufficient heat transfer to maintain computer hardware within operating temperature limits.

USUSat Thermal Subsystem Budget

The mass consumed by the USUSat thermal subsystem is shown in Table 12. As can be seen, the actual thermal subsystem was successfully designed to use substantially less mass than anticipated. However, much of the mass that resulted from the thermally driven redesign of the flight computer enclosure has been included in the structural subsystem.

Table 12. Thermal Subsystem Mass Budget

Subsystem	Component	Predicted Mass (g)	Actual Mass (g)
Thermal	Kapton Strip Heaters	136.0	10.0
	Temp. Monitors	5.0	50.0
	Thermostats	50.0	20.0
	Total	**191.0**	**80.0**

Spacecraft Communications

USUSat Communications

USUSat's communications subsystem may be the system that has changed the most since the original design concept was formalized. These design changes sought to bring additional functionality and to bring in additional funding to the program. The resultant design of the communications subsystem is described below. The system is described in detail in the thesis written by Anuradha Chandrasekharan (2002) and in a conference paper written by the communications team (Chandrasekharan, Gutshall, and Swenson 2001) and so only an overview is given here.

Downlink Communications

The downlink communication subsystem of USUSat was designed with high data rate requirements in mind. Many small spacecraft use downlink rates of around 1200 – 9600 bits per second (bps). USUSat requires a downlink data rate of approximately 100 kbps. This is one to two orders of magnitude larger than required for most small spacecraft and requires transmitters capable of handling the extra data. The transmitter originally selected for use on USUSat was the L3 T-400 transmitter from L3 communications. This transmitter had a variable frequency that could be selected from 2.2 – 2.4 GHz. This frequency is in the military communications band and thus carries with it, the extra requirements of frequency allocation. Program management believed that the extra effort was justified since this transmitter provided the higher data rate required. It was also believed that experience with working in higher frequency communication would be useful for successive projects that also planned to use military band communications. The L3 T-400 is shown in Figure 31.

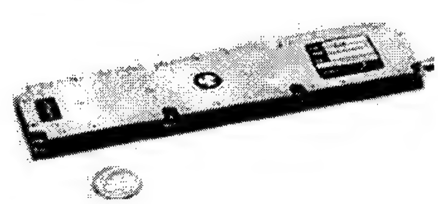


Figure 31. L3 Communications T-400 transmitter.

During the course of system design, communications engineers decided to use another L3 transmitter that gave additional functionality. The new model that was selected was the L3 ST-802S. This transmitter was smaller, had less mass and functioned similarly to the T-400 transmitter. The telemetry stream would come directly from the USUSat flight computer. The data would be formatted into a structure that was similar to the AX.25 protocol data structure.

This allowed engineers to combine real time and stored data into USUSat's telemetry stream, so that system operators on the ground could both receive the stored scientific data generated by the science payload as well as track the spacecraft health in real time. The ST-802S transmitter is shown in Figure 32.

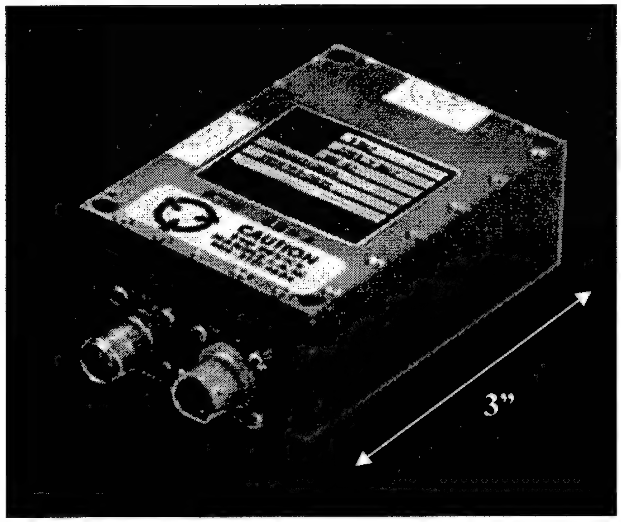


Figure 32. L3 Communications ST-802S transmitter.

Uplink Communications

The preliminary uplink design of USUSat relied on another L3 communications product, the CAR-915A receiver. This receiver was capable of receiving incoming communications from 400 to 470 MHz. It was designed to work with FSK transmissions and to convert them into a data stream for use by the flight computer. This receiver is shown in Figure 33.

During the design of the communications subsystem, the design team decided that they would need to make a change. They found a new receiver that was slightly smaller and consumed slightly less power. Additionally, they decided that the uplink communications system for ION-F needed a terminal node controller (TNC). A TNC serves to accept incoming radio transmissions and to verify them for use by an individual spacecraft. The team believed that during some of the passes over the ground stations, the spacecraft in the ION-F constellation would be proximate enough that more than one spacecraft would be able to receive the same signals. By transmitting identical signals to different spacecraft, undesirable results might be realized.

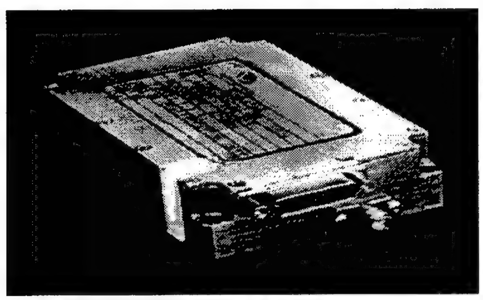


Figure 33. L3 Communications CAR-915A receiver.

In addition, the 3CS constellation was also using uplink frequencies around 450 MHz, just as ION-F had planned. The reason for this choice was that 450 MHz is technically in both the amateur and military spectrums simultaneously. Therefore, equipment designed for military use could access amateur frequencies as well, thus opening a large number of new possibilities for the communications teams. There was, therefore, a chance that ION-F communications could interrupt 3CS operations and vice versa. The TNC functions by receiving incoming transmissions and filtering them for instructions meant specifically for each spacecraft (Gutshall, Chandrasekharan and Swenson 2001).

Unfortunately, a commercial TNC that performed the functions desired by the communications team did not exist so they set out to design their own. By switching to a new receiver, integration of the TNC functions with the receiver would be much simpler. The team therefore decided to use a receiver from Tekk called the Tekk 960L. This also required the team to use a Hamtronics MO 96 modem in order to fully convert incoming signals into a data stream for the flight computer. The Tekk 960L is shown in Figure 34.

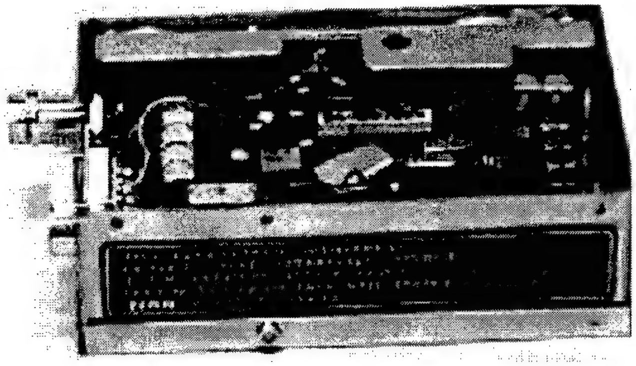


Figure 34. Tekk 960L receiver.

The receiver and modem combo were designed for use on the ground. This meant that the communications team would have to perform several modifications in order to make them spaceworthy. All electrolytic capacitors had to be replaced with tantalum capacitors, and variable components had to be replaced with the necessary fixed value components to produce the desired frequencies. The boards had to be conformally coated in order to prevent outgassing and a new box had to be built to accommodate all of the electronics.

GPS and Telemetry Beacon

One element of the communications subsystem that was not originally part of USUSat is an emergency GPS and telemetry beacon. This beacon is designed to broadcast vital spacecraft information every few seconds in short bursts. This information would include GPS position data, spacecraft temperature, bus and battery voltage and other essential information. It would be broadcast at approximately 145 MHz. This frequency lies within the amateur band and allows USUSat to make these transmissions worldwide. This beacon frequency had been used previously for other applications in ground based systems, but had not been utilized for spacecraft previously. The ground stations are very simple and are utilized by a large number of amateur radio enthusiasts. The data is broadcast in a format that can be automatically streamed onto the internet and would therefore be available worldwide nearly instantaneously. In emergency situations, it would allow operators to receive some data and allow them to try and correct problems.

This system is based on using an APRS MIM 2.0 controller in conjunction with a Hamtronics TA-51 transmitter. Since these systems were originally designed for terrestrial use, the same modifications that were performed for the uplink system also had to be performed on this system. The APRS controller is shown in Figure 35.

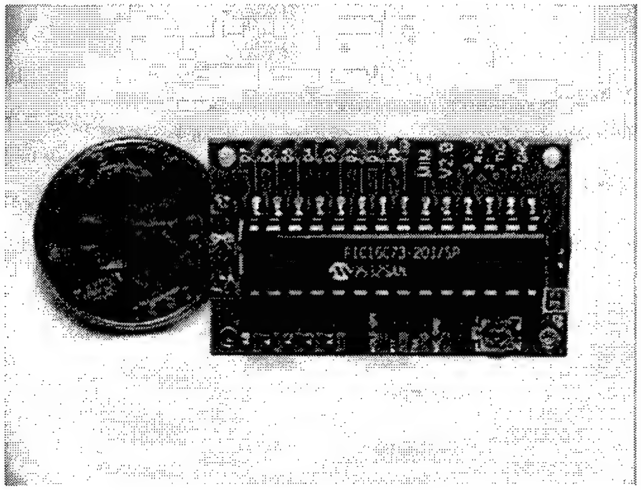


Figure 35. APRS MIM 2.0 beacon controller.

GPS Receiver and Crosslink System

The GPS receiver and crosslink system that will be used on the ION-F constellation was designed by Johns Hopkins University's Applied Physics Laboratory (APL). The system was commissioned by GSFC designers who wished to use this system on future NASA payloads and saw ION-F as a way to create initial flight heritage and test the system in a space flight environment. This system was designed to accurately take GPS measurements at altitudes and speeds for which GPS is normally unavailable. In addition, it was intended to provide interspacecraft communication for the members of the ION-F constellation.

Original specifications described a small system. All crosslink hardware was to have used 0.75 kg of mass and would have fit into an area of 150 x 150 x 100 mm. The system would have used a small patch antenna for GPS reception and would have used fixed monopole antennas for crosslink communication. The system would have to consume no more than 3.5 W of power and average power consumption would be around 2.5 W total.

However, the design that was received from APL was very different from these specifications. At a design review, the system that was presented required at least five components. A main chassis that contained a small computer with transmitter and receiver functions. The design also required external pre-amplifiers for the GPS and received crosslink signal strengths. An isolator and a power switch were also to control transmission and reception through the spacecraft antennas. The total system used about 1.3 kg, took nearly double the original volume, and used about 10 W of total power with average consumption between 4 W and 7 W depending on how often the crosslink functions were utilized. APL designers had originally wanted a system of four antennas with exceptional characteristics but eventually settled on two antennas; a fixed patch for GPS reception and a fixed monopole antenna for crosslink functions. The monopole antenna would be mounted perpendicular to one of USUSat's side panels. This arrangement violated USUSat's envelope on the MSDS but not the SHELS platform so AFRL program management approved the violation.

For a short time, it appeared that the APL system would not be finished within the ION-F mission timeframe and so a backup option was developed by ION-F. This option would have involved the modification of another Tekk 960L similar to the one used in ION-F's uplink communications. A Magellan GPS receiver would have been added as well. Two antennas similar to those specified by APL would have been used. The system was estimated to have used

around 800 g of mass, used two boxes rather than five, and used around 3 W of power on average and 8 W maximum. GSFC engineers were committed to seeing their equipment fly and set aside resources to finish the APL system in time and this backup option was never designed past a conceptual stage.

Antennas

The antennas that are used by USUSat changed a few times during the system design. The changes were generally related to the frequencies that were used in communications and the viability of reception using antennas at these frequencies. For downlink communications at 2.2 GHz, patch antennas were relatively simple to build. Using TMM 10i material from Rogers Microwave, the patches were roughly 1" square. Three of the patches were placed on the spacecraft and a link splitter circuit from Mini Circuits was used to split the signal from the transmitter to the antennas. The patches were placed on three separate surfaces, again due to the formation flying mission objectives. Since USUSat would have to rotate in order to complete formation flying objectives, it was not known what surface would be directed toward the earth during communications overpasses. Using the three antennas would give around 75% coverage if the spacecraft was oriented randomly. Since operators did have some pointing latitude even during formation flying maneuvers, the actual coverage was closer to 95% – 99%.

Uplink antennas were originally envisioned to be fixed monopole tape antennas that would be mounted nearly flat with the spacecraft structure. However, as described in the mechanisms section, communications engineers felt that using dipole antennas would be superior in terms of coverage and performance. Using dipoles meant that deployable antennas were required. Mechanisms designers were unable to produce a design that would satisfy both communications engineers and NASA safety engineers, so a decision was made to switch to an array of patch antennas similar to that used by the downlink subsystem. Due to the 450 MHz frequency for uplink communications, the new patches measured around 4.5" square. They were also made from TMM 10i material and also required a link combiner from Mini Circuits.

The antennas that would be used by the beacon were not originally included in preliminary designs. As explained previously, communications engineers decided that deployable dipoles would be the only feasible method of designing this antenna. However, during a series of program reviews, communications engineers came up with an alternative design that they reasoned would come close in performance to the deployable antennas and would be capable using a fixed antenna. The antenna would be a fixed loop with a 6" diameter. The loop was made of copper and had to be mounted around 1" from any structural backing. This loop was placed on the bottom of USUSat in the center of the Lightband separation system ring. This Lightband ring had a stayout zone only near the edges of the bottom structural panel and it was therefore possible to include solar cells, the fixed beacon ring antenna, one uplink patch antenna, one downlink patch antenna and one camera used in the ADCS subsystem. The layout of components on the USUSat bottom panel is shown in Figure 36.

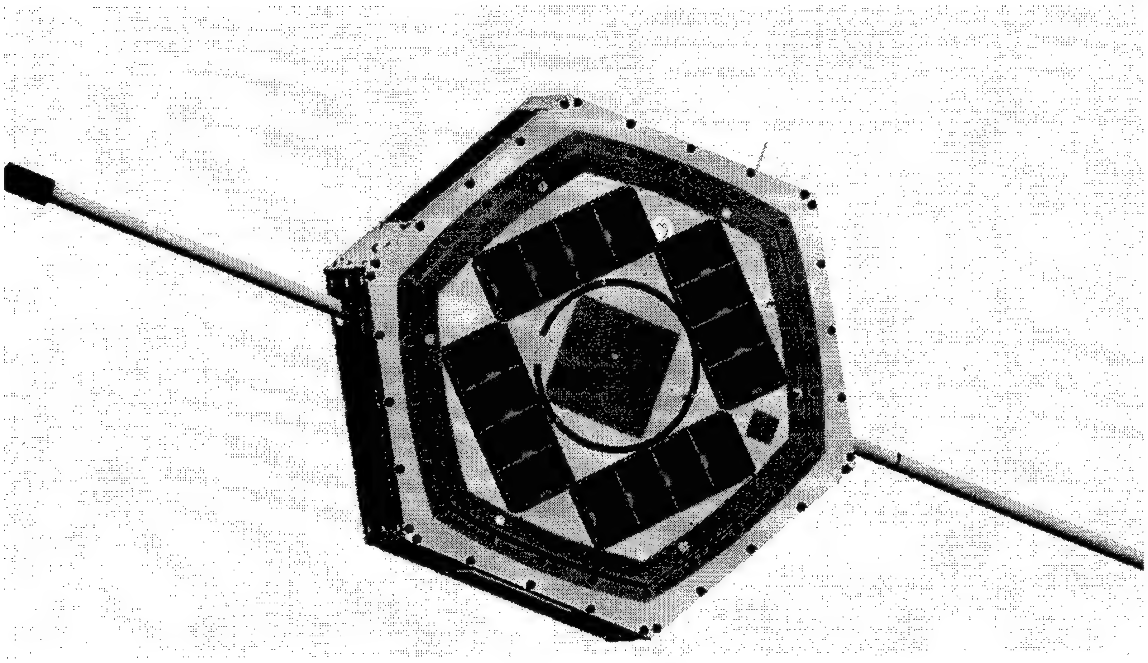


Figure 36. Bottom panel external components.

The last two antennas that were required were for the GPS and crosslink subsystem. A small patch antenna from Toko was placed on the panel that would spend the most time in the zenith orientation in order to maximize GPS signal reception. The antenna that would be used for crosslink communications was designed by Virginia Tech. This antenna was a fixed monopole that had to extend either from the nominal nadir or zenith pointing panels. The antenna was fabricated from a small copper tube that was soldered to a SMA connector. This connector was then attached to a hexagonal aluminum base. USUSat chose to mount this antenna to its nominal nadir face near one of its uplink patch antennas.

USUSat Communications System Budget

The communications subsystem required several iterations before engineers were satisfied that they had produced a system that could reliably complete its objectives. The final system included several components that were not envisioned in preliminary designs and some extra mass and power were consumed by the system. A mass budget for the communications subsystem is shown in Table 13.

As can be seen, some extra mass was reallocated to the communications subsystem. One interesting observation is that most of the major components were designed with less mass than anticipated. It was the components that were unaccounted for in initial budgets that drove the mass increase. One major savings was in the design of the data formatter. Originally conceived as a separate part that would package data from the flight computer into a form usable by the transmitter, it was redesigned as a modular part of the flight computer electronics. The design that allowed for the mass reduction is described later. In summary, although the communications subsystem exceeded its original budget, it was believed that the additional capacity was worth the extra mass and power.

Table 13. Communications Subsystem Mass Budget

Subsystem	Component	Predicted Mass (g)	Actual Mass (g)
Communications	GPS Receiver	680.0	0.0
	S-Band Transmitter	0.0	203.0
	Receiver	282.0	232.0
	Beacon Transmitter	0.0	383.0
	Link Matching Circuits	0.0	178.0
	Data Formatter	907.0	172.0
	Crosslink / GPS	454.0	1292.0
	Antennas	0.0	349.0
	Total	**2323.0**	**2460.0**

Spacecraft Command and Data Handling

USUSat C&DH Design

The flight computer or C&DH system is the brains of the spacecraft and was designed to be so for USUSat. The flight computer is responsible for interfacing and commanding all other subsystems. It is responsible for processing all data and commands and has been designed to operate as autonomously as possible. The design of the USUSat C&DH subsystem is detailed in the thesis written by John Jensen (2000) and in a conference paper by Barjatya, Nelsen, Swenson and Fish (2002). Following is the design summary.

The C&DH subsystem originally considered using a processor and board combo called the Tattletale 8. This system represented a tested design that had an extensive library of software and students had previous experience with the system. Unfortunately, it had limited ability for expansion and addressing multiple peripherals. The system was also incapable of booting from external memory. C&DH engineers decided to design a custom system that would include a large amount of flash memory for data storage and that would be modular enough to address the different peripherals that the spacecraft in ION-F constellation desired. It was also decided to use the VxWorks real time operating system so engineers looked for a chip that was compatible with VxWorks and support the external operations that were required.

ION-F originally looked at Sharp, Motorola and Hitachi processors before finally selecting a Hitachi SH-7709 processor. This processor was capable of around 75 million instructions per second which allowed designers to design the software that would allow ION-F spacecraft to operate autonomously. ION-F decided to use a modular design that would allow several different boards to interface with the main flight computer. These boards could be designed to interface with the particular hardware on each spacecraft. The modular box design that was used for ION-F's computer system is shown in Figure 37.

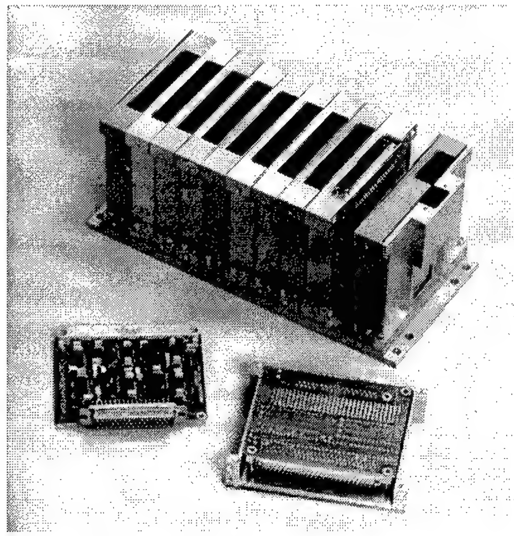


Figure 37. Modular design of electronics enclosure.

This design was originally used by SDL engineers who wanted a modular electronics enclosure for sounding rocket missions. In this design, a backplane runs down the length of the box with traces etched onto it to allow for the transmission of electrical signals. Located above this backplane are a series of electronic cards that contain the components that actually perform the desired functions. Each card interfaces with the backplane through a common connector. Each card is surrounded by an aluminum bracket that stabilizes the cards for launch and transfers heat out of them during operation. The box and backplane can be lengthened or shortened for different missions and different cards can be interfaced. In ION-F's case, designers chose to use ten cards, some of which would be common and some of which would be tailored to the needs of individual universities. The allocation of cards within USUSat is shown in Figure 38.

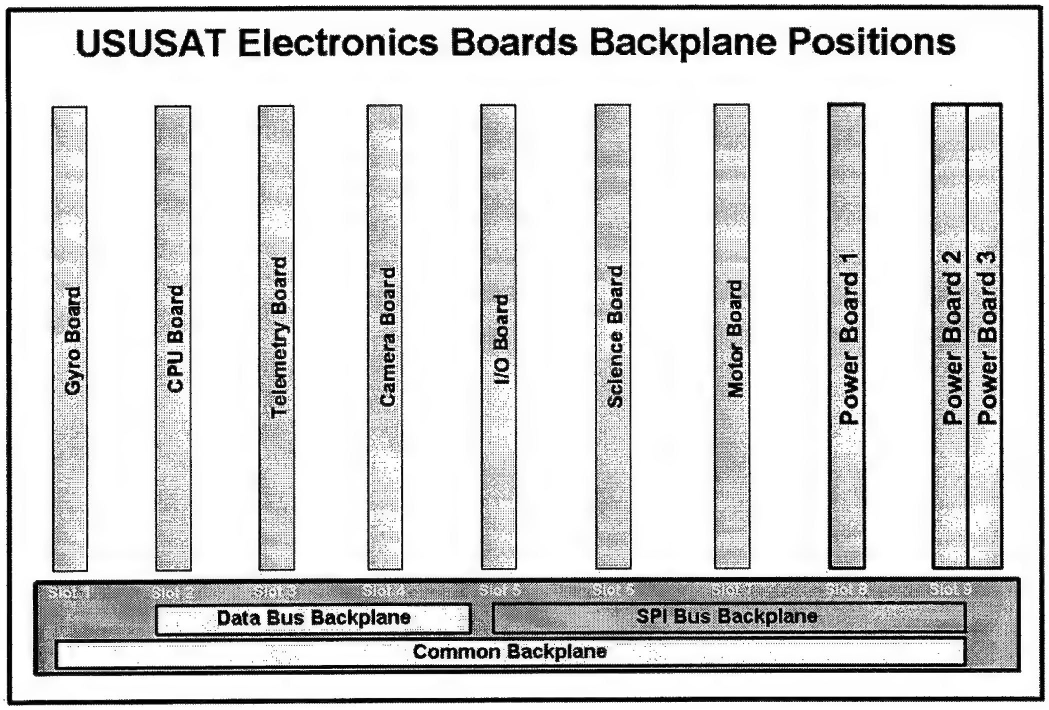


Figure 38. USUSat backplane and electronic card definition.

The first six boards in each spacecraft would be identical and the last four could be tailored as each school saw fit. The first card contained electronics that interfaced with the inertial rate gyros used for the ADCS subsystem. The gyros would be mounted mechanically on the end of the CEE in order to keep wire length as short as possible. The CPU card contained the Hitachi microprocessor and most of the electronics necessary to interface with it including the boot ROM and scratch RAM. The telemetry card served to format the data for use by the downlink transmitter. It also contained the majority of the system memory in order to store for transmission. The camera board contained the electronics necessary to interface with the CMOS cameras used by the ADCS subsystem. The I/O interface card contained the analog to digital (A/D) converters and other hardware necessary to convert the signals coming in from sensors around the spacecraft into digital signals to be used by the flight computer. The science card contained the electronics needed to process incoming science information. The motor control card contained the necessary electronics for driving the stepper motors used by the magnetic gimbal. The last three boards contained power system electronics.

This system was designed with its own base and card slots, but as described previously, was redesigned so that the base of the CEE was integral to the structure of USUSat. For reasons discussed later, the gyros and gyro board were not required and therefore removed to save mass and power.

The system was designed so that several mission operating modes would be recognized and it respond accordingly. A ground mode in which tests were being run would be indicated by connection to the system being made through the USUSat auxiliary port. In this mode the spacecraft would essentially idle unless operators requested some operation through ground support equipment (GSE). A stack mode in which the spacecraft was still connected to the rest

of the ION-F formation was designed so that USUSat could perform initial checks of spacecraft hardware to try and ensure that systems were operating normally and so that batteries could be charged. In this mode, USUSat would wait until its second set of relays was released at which time it would initiate separation of the ION-F stack and the deployment of its booms. A fault mode was included so that if the spacecraft rebooted and was not on the ground or in the stack it would know that something had gone wrong. It could check sensors to see if some known fault had occurred such as low power or overheating in some area. If power was low, the spacecraft could initiate a sun-pointing mode until the batteries were recharged. If overheating was detected, the spacecraft could turn the affected area away from the sun and attempt to shut off power to the affected subsystem. Faults due to SEUs or SELs would have been corrected by power cycling the system and automatic restarts from the power system would alleviate the problems (Jensen and Swenson 2000).

Once any faults were corrected, the spacecraft could proceed with executing its master schedule. This schedule could insert the spacecraft into a solo mode where it would collect scientific data and try to maximize solar energy collection. A formation flying mode in which the spacecraft would try to coordinate its activities with the other spacecraft from the ION-F constellation is also available. An orbit maneuvering mode could also be entered. This mode would attempt to use the drag modulation techniques in order to change parameters of the spacecraft's orbit other than the altitude. Finally, a ground communication mode is also available. This mode can only be entered due to the reception of a signal from one of ION-F's groundstations. International law prevents transmitting without the authorization of the country it is flying over due to the potential for interference with other existing programs. Therefore, if and only if, the spacecraft is receiving signals from its groundstation can it transmit any information. The exception to this rule is the emergency beacon. Since it transmits in amateur frequencies and transmits data that is identical in format to data already transmitted at that frequency, it can transmit continually. A diagram showing the USUSat software modes is shown in Figure 39.

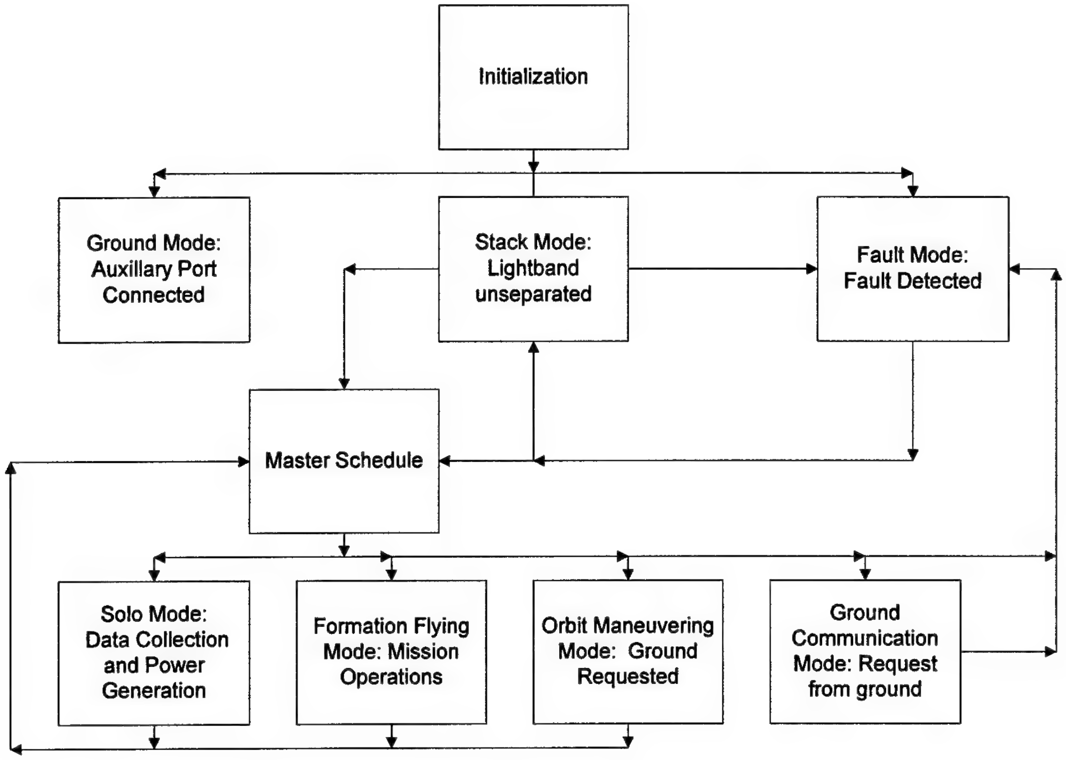


Figure 39. USUSat software master sequence.

USUSat's computer system used industrial grade electronics. Therefore, it was necessary to use circuit design techniques in order to guard against radiation effects since industrial parts are not hardened against radiation. Since the initial boot up code had to be protected, it received the only rad hardened component in the CPU. A 256 KB EEPROM was used to store initial code so that the system could always boot up reliably to a known configuration. Eight MB of flash memory in a triply redundant voting logic configuration could be used to store essential information such as uploaded code updates or new portions of mission schedules. As stated, a triple redundant voting logic scheme was used to prevent SEUs from affecting important code. The voting scheme works by maintaining three copies of any important code. If any discrepancy is detected between the three, the C&DH subsystem finds the two identical copies and replaces the third, corrupted copy with a new copy of the uncorrupted data. Five MB of SRAM was then used as a system scratchpad for calculations. Since this data was needed for only a short time, any bit errors in the data could be ignored. To guard against latch-up, the computer had redundant watchdog timers. These timers had to be reset by computer command or they would power cycle the flight computer to try to eliminate the latch-up. Latch-ups could also be detected by high current draw in the power system. If the power system detected abnormally high current draw in some electronics, it would attempt to power cycle the hardware. If the affected hardware was permanently affected, the power system would permanently disable power flow to that subsystem. The flight computer also had sensors that monitored the power system. If the flight computer detected latch-up within the power system, it would power cycle the entire spacecraft to deal with the problem. A functional diagram of the C&DH system, including the features discussed above, is shown in Figure 40.

One important note to include that would have saved much effort deals with the compatibility of software and hardware. One reason that the Hitachi SH-7709 was selected was because it was listed as being compatible with the VxWorks operating system. After numerous tests showed problem after problem, Wind Rivers – VxWorks' designer – was contacted to see if they could offer ideas. The SH-7709 was listed as being compatible, but was compatible with a specific set of hardware. By introducing external hardware that had not been previously tested, ION-F was forced to write new interface software in order to get the VxWorks OS just to boot up and execute on the ION-F flight computer. It is often helpful to find out exactly what is implied in hardware and software compatibility charts.

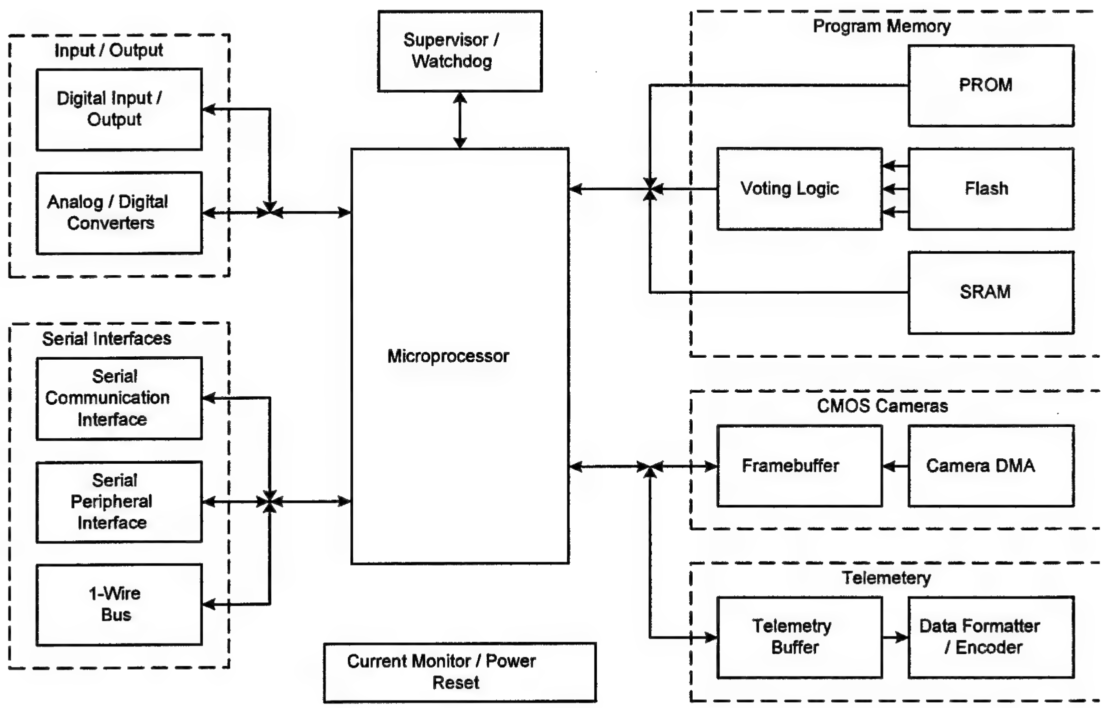


Figure 40. USUSat C&DH architecture.

USUSat C&DH Subsystem Budget

The C&DH subsystem took substantially longer to design due to this and other issues. Looking again at Table 11, it can be seen that the flight computer system as designed used slightly more power than was anticipated. Since a custom design was used, it is difficult to make accurate preliminary estimates about power consumption. The design as completed represents the power required for necessary functions and operations. The mass allocated to the C&DH subsystem is shown in Table 14.

In this table, it is possible to see one subsystem for which preliminary budgets were unrealistic. The actual electronics components that compose the flight computer and I/O or data buffer have masses very close to the initial budgets; however no mass was originally allocated for the housing or cards that the components would have to be mounted on. The shielding was originally conceived as a small amount of aluminum that could be placed around the computer to help protect against SEUs and SELs. The original budget neglected mounting provision and heat transfer provision. The final design balanced minimum mass against vibration resistance and heat transfer requirements. The original budget in this case was unrealistic.

Table 14. C&DH Subsystem Mass Budget

Subsystem	Component	Predicted Mass (g)	Actual Mass (g)
C&DH	Flight Computer	30.0	105.0
	Data Buffer	55.0	158.0
	Shielding	90.0	779.0
	Total	**175.0**	**1042.0**

Spacecraft Attitude Determination and Control

USUSat Attitude Determination and Control

The ADCS subsystem was designed to provide accurate knowledge and control over the attitude of the spacecraft to support the formation flying mission of the ION-F constellation (Humphreys, Fullmer, and Swenson 2002). The system uses four Complementary Metal-Oxide-Semiconductor (CMOS) cameras that can be used as both horizon and sun sensors for fine attitude determination. A three axis fluxgate magnetometer is also included on the tip of one of USUSat's deployable booms as discussed previously. In addition, the ADCS subsystem can also use the voltage values from the spacecraft solar panels to obtain a rough estimate for spacecraft attitude (Meller, Sripruetkiat, and Makovec 2000).

<u>Cameras</u>

The cameras that will be used by the ION-F are Fuga Model 15d CMOS cameras. These cameras are black and white cameras that have a resolution of 512 x 512 pixels. The camera uses eight bits for grayscale color description. These cameras have been used previously on missions conducted by the European Space Agency (ESA) to verify mechanism deployment. The cameras require the use of a system of lenses from Edmonds Scientific. The Fuga 15d camera is shown in Figure 41.

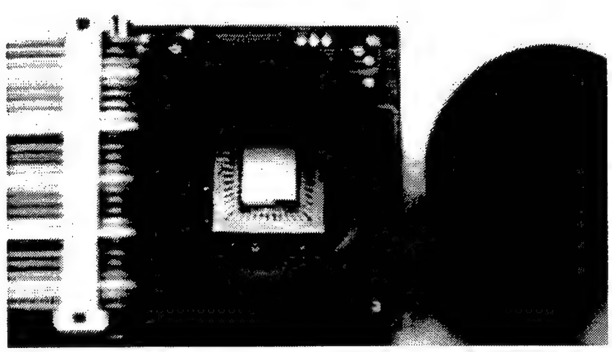


Figure 41. Fuga 15d CMOS camera.

These cameras behave similar to 256 KB ROM chips. Software can access an x and y position in the sensor and directly read out as an eight bit word. The camera outputs these words

as a direct measure of photocurrent and as such does not require an integral amount of time to elapse between readings. Using iris lenses from Edmonds Scientific allows each camera to have a field of view (FOV) of around 67°. USUSat uses four of these cameras in order to maximize coverage for the spacecraft.

Each camera has to have supporting equipment. Each camera requires lenses and barrels to fix the lenses in a fixed position. NASA safety also requires that all of the glass in the lens configuration to be completely contained in case they shatter under launch vibration. Each camera card is then mounted to an aluminum barrel. A small aluminum plate fixes the camera to the structure and a clear, shatterproof, thermoplastic optical window is placed over the plate. A small retaining clip holds the optical window in place. The camera mounting hardware is shown in Figure 42.

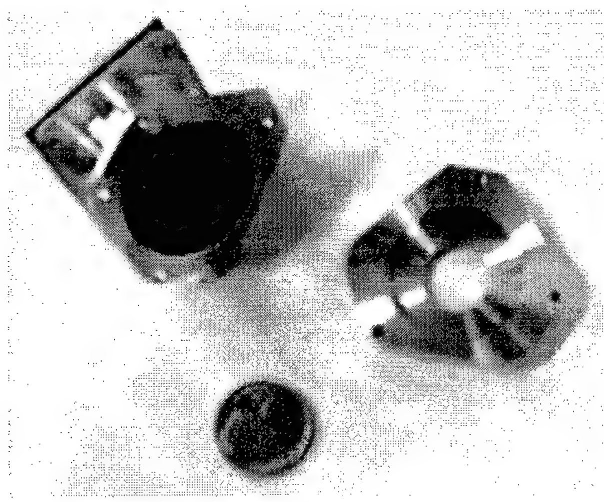


Figure 42. Camera mounting hardware.

The algorithms used to process the images are based on one dimensional edge detection algorithms. These algorithms use three scans per image to detect three horizon positions and extrapolate a sun pointing and earth pointing vector than can be used to establish the spacecraft attitude. The algorithms are capable of dealing with such situations as terminators on the earth, the moon in the FOV, and glints or reflections from other spacecraft. The ADCS team has estimated that the algorithms used are capable of determining the spacecraft attitude to within ± 3°.

Four cameras were originally included in the USUSat design as horizon sensors only. USUSat would originally have used a separate set of dedicated sun sensors that were being designed by USU students. However, preliminary designs were not promising. When the software engineers thought that the cameras could be used for sun sensing as well, the dedicated sun sensors were eliminated from the design. The ADCS team wanted eight cameras however to ensure a large amount of coverage. Unfortunately there was insufficient volume to accommodate eight cameras. In addition, the interface electronics designed to accommodate the cameras were to be common among all three universities in ION-F. UW and VT were using only four cameras and therefore the electronics could not be expanded to accommodate eight cameras so the final USUSat design uses four cameras for horizon and sun sensing.

Magnetometer

The magnetometer that will be used on USUSat is built by Applied Physics Systems and is the APS-533 model. This model is a three axis fluxgate magnetometer. It is a small magnetometer, roughly the size of a C or D cell battery and has a mass of roughly 18 grams. The magnetometer produces three analog outputs that can be used to detect the magnitude and direction of the earth's magnetic field. This sensor can be used on a time sampled basis in order to give a good estimate of the body rotation rates of the spacecraft. The APS-533 magnetometer is shown in Figure 43.

Solar Panel Estimation

USUSat ADCS engineers can use the normalized inputs from the solar panel voltages to gain a rough estimate of the current sun vector. This can help simplify operations in the sun and horizon sensors by giving a preliminary estimate of which cameras should be able to see the sun within their FOV. In the case of failures of the cameras, the solar panels can help to give a very coarse estimate of the sun and nadir vectors. These estimates can also be used to help eliminate false horizons when using the cameras for horizon detection.

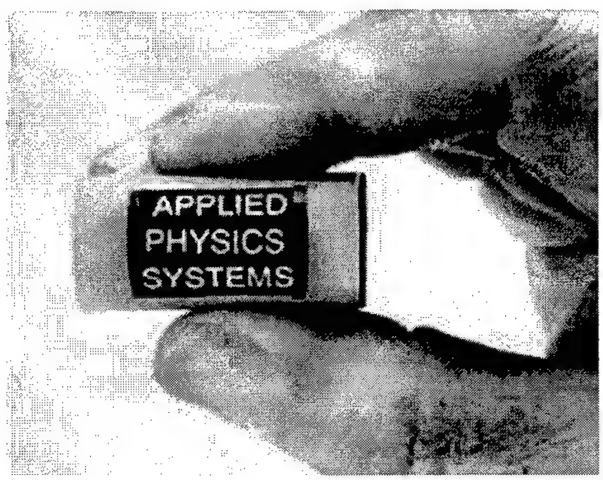


Figure 43. APS-533 three-axis fluxgate magnetometer.

Attitude Determination Methods

The software methods that ADCS engineers use to actually determine a reliable estimate of spacecraft attitude are some of the most complicated software routines on the spacecraft. The flight computer must be able to process digital images, read magnetometer data, read solar panel data, integrate all this data into an attitude estimate, use a filtering method to ensure that accurate estimates are obtained, propagate the spacecraft position in orbit, and compare this position to data either uploaded by operators or obtained from the GPS receiver. The overall process used for attitude estimation is shown in Figure 44.

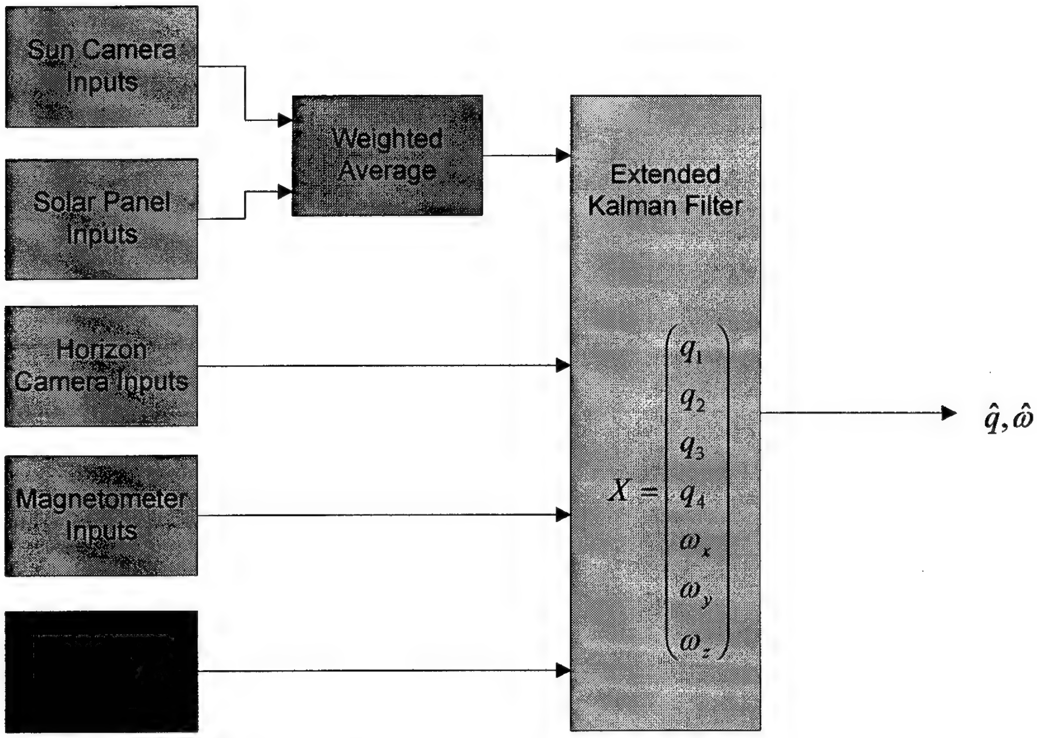


Figure 44. Attitude estimation algorithm.

The process shown here allows the spacecraft to estimate its attitude and rotation rates with good accuracy. Sun camera and panel inputs are averaged and then combined with horizon camera and magnetometer data. Sensor logic converts their readings into estimated sun and nadir vectors. These inputs are combined into an extended Kalman filter. The filter does the job of comparing the inputs with previous attitudes. It then filters the inputs to produce its "best" estimate of the spacecrafts current attitude information. This information can then be used in the control algorithms that determine spacecraft commands. The algorithm shown above is estimated to provide attitude information to within 1-10° of accuracy.

The accuracy can fluctuate with this algorithm because sun vector and horizon data are unavailable while the spacecraft is eclipsed. In addition to this problem, the software used by the cameras is new and untested. It works well in laboratory conditions, but actual flight performance is unknown. With the small number of sensors employed, the Kalman filter takes some time to converge to a solution and the body rate estimates are somewhat noisy.

Inertial Rate Gyros

Initial designs also called for the addition of inertial rate gyros. QRS-11 rate gyros from Systron Donner were investigated and purchased. The University of Washington was responsible for designing the electronics that would interface with the gyros. The use of gyros would have meant that much more accurate information would be available for ADCS engineers. However, they also posed design problems. The gyros would have added an around 4.75 W of additional power consumption as well as 550 g of mass. The larger problem came in the software that would have been used. In order to incorporate the rate gyros, a separate Kalman filter routine would have to be written. Then, since USUSat engineers anticipated power cycling the gyros to reduce average power consumption, ADCS engineers would have to write code that could alternate between the two filtering routines. Significant testing would have to be undertaken to ensure that the switching occurred properly. Due to these factors, the decision was made to remove the rate gyros from the design.

ADCS System Simulation

In order to accurately predict performance, a software simulation testbed was developed by four key members of the USUSat ADCS team: Todd Humphreys, Angela Millsap, Prapat Sripruetkiat, and Jinsong Liang. A simulation was developed in the Matlab / Simulink environment that would model the spacecraft dynamics and external environment. The simulation would then predict sensor responses to external environment. Estimates would then be generated for spacecraft attitude and control responses would be generated. The front end of the attitude simulator is shown in Figure 45.

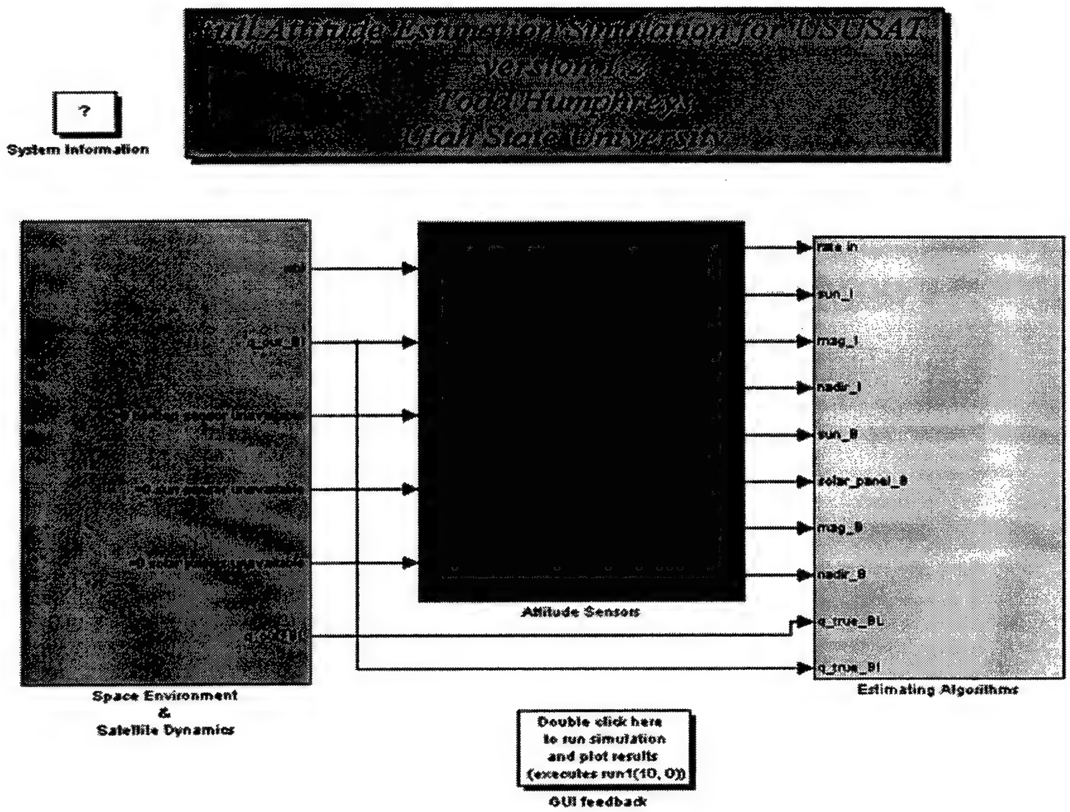


Figure 45. USUSat attitude simulation front end.

The work that went into developing this simulation was split up between the four members of the ADCS team. Todd Humphreys was responsible for developing the Kalman filter, the code to generate solar panel attitude estimates, magnetometer data, and to simulate the space environment. He was also responsible for integrating the work done by the other team members. Prapat Sripruetkiat was responsible for generating the algorithms that performed the sun and horizon sensor data. Angela Millsap was responsible for modeling the drag effects and the formation flying navigational software. Jinsong Liang was responsible for generating the code that took the attitude information and determined the appropriate control responses that would be sent to the gimbal to complete the navigation requests.

The software was complex, but on average, attitude knowledge was generated every five seconds. New navigation commands would be issued approximately every ten minutes and control commands would be issued as necessary. Spacecraft software would also include magnetic field models and an orbital propagator that would be used. The orbital propagator could also be supplemented by data from the GPS receiver and by ephemeris data transmitted from the ground if necessary.

One important realization that occurred during the development of software was that the Hitachi SH-7709 processor did not have a floating point unit (FPU). The lack of an FPU meant that any floating point operations would have to be simulated by the processor using integer based mathematics. This would significantly slow operations since most of the ADCS software

had been designed to use double precision mathematics. Much of the ADCS software should be rewritten to use integer based mathematics instead. Only absolutely essential calculations would be allowed to use floating point mathematics.

USUSat ADCS Subsystem Budget

Table 15 documents the mass consumption in the ADCS subsystem as designed. The ADCS design team came very close to meeting their original design budget. The cameras came in using less mass than anticipated. The control and camera interface electronics appear to have been neglected in the preliminary design. It was originally believed that these systems could be interfaced through the I/O board included with the C&DH subsystem, but during the design, it became apparent that dedicated electronics would be necessary to interface with these subsystems. Preliminary designs also called for the possibility of torque coils. Simulations with the gimbal indicated that it would be sufficient for control and the coils were eliminated. The rate gyros were also eliminated for the reasons discussed above.

Table 15. ADCS Subsystem Mass Budget

Subsystem	Component	Predicted Mass (g)	Actual Mass (g)
ADCS	CMOS Camera	400.0	372.0
	Magnetometer	50.0	20.0
	Sun Sensor	600.0	0.0
	Camera Electronics	0.0	167.0
	Control Electronics	0.0	244.0
	Torquer Coils	181.0	0.0
	Rate Sensors	91.0	0.0
	Total	**1231.0**	**803.0**

USUSat Science Payload

The main science experimentation flown on the ION-F constellation is a pair of probe antennas that work the measure electron density and plasma frequency in the ionosphere. This research is of interest since the behavior of ionosphere affects the propagation of radio signals. As our society depends more on satellites for communications, navigation, and geolocation, better knowledge of ionospheric behavior is necessary to design better systems to accomplish these goals.

Some experiments have been conducted using sounding rocket payloads or using individual spacecraft. However, these tests do not allow experimenters to collect data on how the ionosphere evolves over time. Since ION-F has three spacecraft in a constellation, it is possible to take measurements of how ionospheric plasma evolves temporally. ION-F would be the first spacecraft constellation to make these systematic measurements as a group.

The science instrumentation that will be flown on the ION-F constellation was designed at SDL and is similar to instrumentation that has flown on previous payloads. USUSat has three main pieces of scientific equipment. The first is the deployable science boom. This boom acts as a plasma interference probe (PIP) and helps take measurements on plasma frequency, electron

density, and electron behavior in the ionosphere. A second piece of instrumentation is a small patch antenna called a direct current patch (DCP). This patch helps provide relative electron density measurements. The last piece of equipment is the electronics required to convert measurements into data.

The equipment on ION-F is intended to complete three major objectives. The first is to document the evolution of plasma structure and ionospheric irregularities. The second is to help determine the spectral characteristics of ionospheric plasma. The third is to develop a global map of the distribution of plasma structures and irregularities.

Science objectives had also called for measurements to be made of GPS signal strength. These measurements could be compared with theoretical values to give further indications of plasma behavior. Due to lack of time and funding, this objective was dropped from the ION-F mission.

USUSat Science Subsystem Budget

Table 16 shows the mass of the science payload for USUSat. The science subsystem as designed used less mass than their original budget. The science electronics and DC probe masses are very close to their initial budget. The science boom has been included in the mechanisms subsystem budget rather than being repeated here.

Table 16. Science Subsystem Mass Budget

Subsystem	Component	Predicted Mass (g)	Actual Mass (g)
Science	Plasma Probe	227.0	20.0
	Science Electronics	227.0	216.0
	Total	**454.0**	**236.0**

CHAPTER 4: SAFETY ENGINEERING IN NASA APPLICATIONS

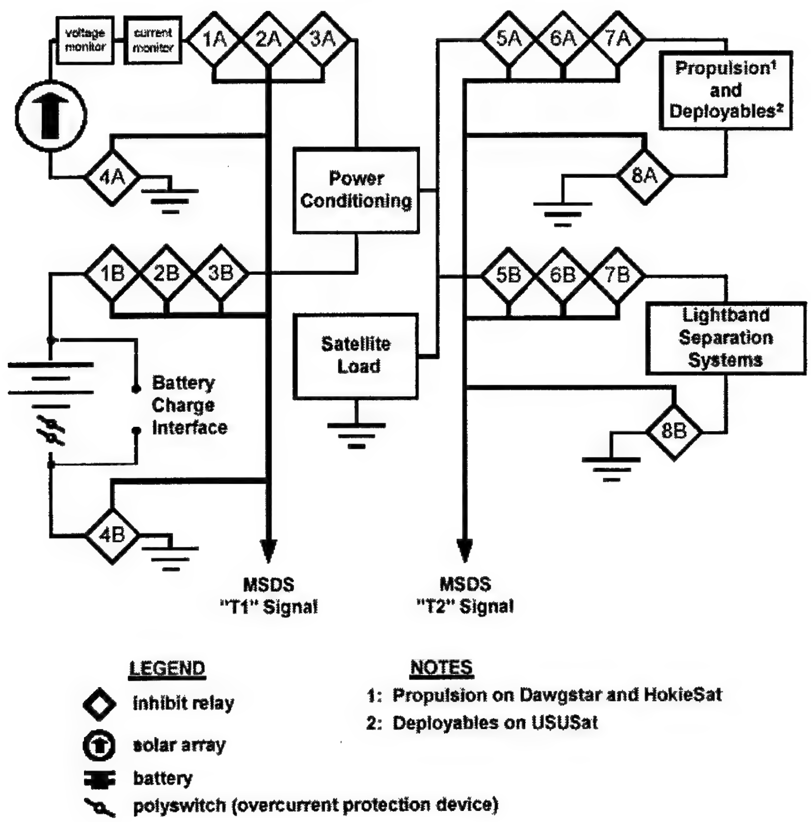
System Safety Engineering

USUSat Electrical Inhibit Scheme

Since ION-F is a small program, designers chose to make use of the unpowered bus exception. By using this exception, payload designers could eliminate most of the hazards associated with electrical systems by ensuring that they would not activate inadvertently. The ION-F inhibit scheme relies upon the MSDS platform designed by AFRL.

In this scheme double pole double throw (DPDT) magnetically latching relays will be used to prevent inadvertent power flow. Four relays will be placed into the circuit between the batteries or solar cells and the spacecraft bus. Since they are double pole relays, the solar cell lines will be run through one pole of each relay while battery power lines are routed through the second pole. Three relays will be placed on the hot side while one is placed on the ground side of the bus. This arrangement is due to NASA safety requirements, which call for four independent inhibits against inadvertent power flow. In addition, four relays are also placed into the lines between the normal spacecraft bus and the Lightband separation system. This arrangement is shown in Figure 46. Using two sets of relays allows them to be triggered independently. The relays cannot be triggered by the spacecraft, but must be triggered by a power signal from the MSDS that houses the stacks within the Shuttle. The relays are also capable of being triggered by a signal from ground support equipment through a connector on the top of USUSat. A schematic of each individual relay is shown in Figure 47.

Attachment 2
ION-F Inhibit Scheme



ION-F Inhibit Scheme
Applicable to all three ION-F nanosats

Inhibits Removed	Signal to Remove	Time Removed	Functions Enabled
1 – 4	MSDS T1 Signal	20 Minutes After SHELS Ejection	Battery Power and Solar Power to all Functions except Lightband, PPTs, and deployable antennas and booms. Solar Charging of Batteries
5 – 8	MSDS T2 Signal	4 Days After SHELS Ejection, Post Orbiter Landing	Battery and Solar Power to Lightband, PPTs, and deployable antennas and booms.

ION-F Inhibit Removal Sequence

Figure 46. ION-F electrical inhibit scheme.

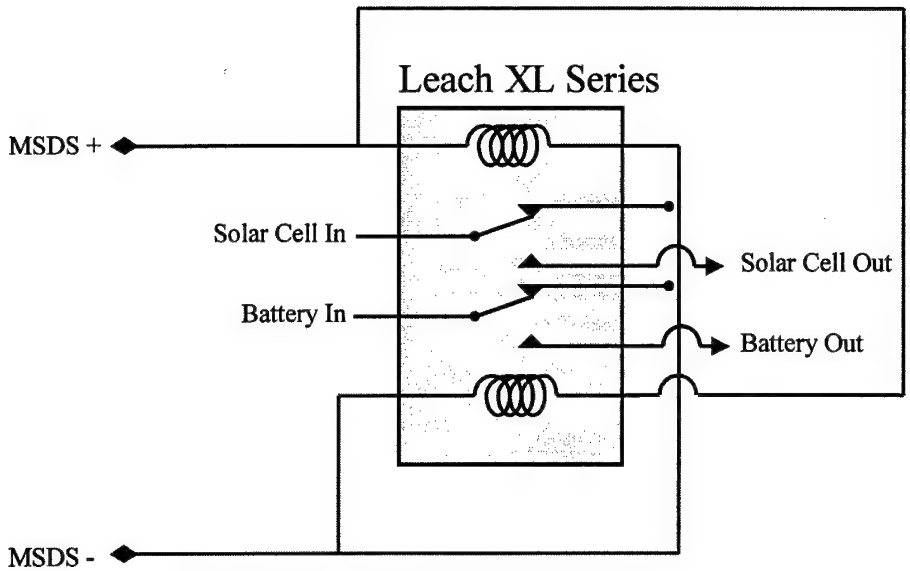


Figure 47. Individual relay schematic.

As can be seen, each relay has two electromagnetic coils. The tip of each relay has a small magnet that causes it to be attracted to the poles of the relay. This small magnet holds the relay in place during vibration allowing it to retain its position during launch. To toggle the relay position, the coil opposite the relay position is activated causing a magnetic field that draws the relay to the opposite pole. By activating the coil closest to the latch with reverse polarity, the coil pushes the magnet away from it causing a similar effect. These relays are single point failures in that if any single relay did not trigger correctly, no power would ever flow through it. In an attempt to mitigate some risk, the incoming MSDS power signal is designed to energize both coils with reverse polarity. This will cause one to push and one to pull simultaneously. Thus, if the relay should fail due to one coil being damaged, the other coil should activate the relay, providing some redundancy. In order to reset the relays, a signal from ground support equipment with opposite polarity will be passed through the coils.

To explain why USUSat needs to have relays that can be triggered at separate times, it will be useful to examine the ION-F mission launch profile shown in Figure 48. As discussed previously, ION-F will be launched from the Space Shuttle. In Figure 48, one can see the ION-F stack and the 3CS stack on the MSDS and SHELS systems in the shuttle bay. ION-F will not be allowed to power up during this phase. However, twenty minutes after the MSDS separates from the Shuttle, it will send the first signal allowing ION-F to power on systems that have been deemed non-hazardous. NASA safety engineers will not allow ION-F and 3CS to separate until the Shuttle has landed. At the time that the Orbiter has landed, the MSDS will send the second power pulse to ION-F and 3CS, and then initiate the separation system that holds these two stacks in place. This allows the spacecraft to activate the separation systems, propulsion systems, and deployables and begin mission operations. Each of the two sets of relays is dedicated to a specific phase of the satellite operations so that full operation is not commenced until the Orbiter has safely landed.

Another requirement for using the unpowered bus exception is that the controls that will activate the inhibits must also be unpowered. The MSDS system relies on a system of microswitches in the separation plane of the SHELS system. These switches interrupt power flow within the MSDS EPS and cannot be removed until the SHELS system has separated from the Shuttle. This separation will energize the MSDS but will not automatically transition the ION-F relays. A set of timers within the MSDS will activate thus allowing time for the MSDS to drift away from the Shuttle. The first timer will activate ION-F non-hazardous functions, while the second will activate the ION-F separation systems.

Figure 48. ION-F mission profile.

Payload Safety – RF Energy

Electromagnetic radiation from transmitting payloads is a cause for concern since this radiation could cause critical errors within mission critical systems and other payloads if large amounts of RF energy are emitted. The Shuttle Interface Control Document (ICD) details the allowable levels of radiation that can be emitted from the payload.

If the spacecraft levels are within 12 dB under the limit, two inhibits are required against premature activation. If the spacecraft levels are greater than the ICD limit, three inhibits are required. If the spacecraft transmission levels are more than 6 dB greater than the allowable limit, three inhibits are required, two of which must be monitored. In addition, all radiation

emitters must include a means of turning the transmitter on or off from either a ground station or the flight deck in case the payload causes unforeseen interference problems.

The unauthorized transmission of RF energy is of major concern because the energy can cause major hazards. RF energy is capable of igniting gases discharged from spacecraft batteries. The energy also has the potential to interfere with Shuttle operations such as communications or pyrotechnic firings. The energy can also cause similar problems in other payloads within the Orbiter. Finally, this energy can interfere with some types of inhibits used in other systems. This interference could remove inhibits that control critical or catastrophic functions.

To verify that RF inhibits are set, test, analysis, inspection, or operational verification through indicators can also be used. Passive verification before the start of the mission is greatly preferred to active indicators. Any procedures that are necessary to verify RF inhibits must be detailed for ground support personnel and flight crew usage.

ION-F communications systems can transmit approximately 2 W of power. This can cause an electrical field intensity of around 149.8 $dB_{\mu}\ {}^{Volts}\!/_{meter}$ in the shuttle payload bay. This amount exceeds the intentional and unintentional transmission limits and is therefore considered to be a catastrophic hazard. Therefore, these systems have been inhibited using the ION-F electrical inhibit strategy discussed previously. USUSat will not be allowed to activate its communications subsystem until after it has been ejected from the Shuttle payload bay.

Payload Safety – Batteries

USUSat Battery Safety

As stated previously, USUSat has selected to use a set of Sanyo NiMH batteries. These batteries were selected for two main reasons. First, they had a large energy density and allowed a large margin for power storage. Second, the cells had been used previously on Shuttle missions in the REBA and EHIP devices. These devices were small portable gear used by astronauts during EVA.

The battery pack itself would be welded together with two polyswitches into a pack. These polyswitches act like resettable fuses. If shorting is drawing large amounts of current, the switches will overheat and trip, interrupting current flow. As the switches cool, they reset and allow current to flow again. If shorting was due to a SEU, this should cycle the spacecraft and allow normal operation to resume. If the shorting is due to a physical defect, the switches will continue to trip and the spacecraft will eventually run out of power.

The pack with the switches would then be surrounded in Pigmat, an absorbent form of polypropylene. This layer of potting would soak up any electrolyte leaks from the pack. The pack would be shrink wrapped in Teflon to maintain its configuration. It would then be placed into an aluminum box. The box would have both electroplated nickel and solethane coatings in order to prevent corrosive effects and to prevent shorting. Two Delrin retainers would fix the pack inside the box to prevent the pack from vibrating.

Each battery cell is individually vented to prevent the buildup of gases within the cells. In addition, two vents were located on the box. These vents were located so that they would either be pointing upward or would be on the top half of the box during any orientation encountered during integration and launch with the Shuttle. The vents were constructed of

porous Teflon. This allowed gases to permeate the vent but trapped any liquids within the box. In this way, a flammable concentration of vent gases could be avoided, but corrosive electrolytes could not escape in the event of a leak that was not fully absorbed by the potting.

Payload Safety – Structures, Fracture Control and Fasteners

USUSat Structural Safety and Fracture Control

USUSat's structure was designed to withstand the launch loads with large margins of safety. As stated previously, structures must maintain factors of safety of 1.25 on yield and 1.4 on ultimate strength if verified by test. AFRL had established the requirement that the UN2 hardware would be verified by vibration testing. To establish the proper safety factors, they took the ± 11 g requirement from the shuttle and found an orthogonal load equivalent since the acceleration could be applied in three axes simultaneously. They then added the required 1.25 factor of safety and arrived at a value of 24 g acceleration loading. AFRL declared that their test would be run at this level and that in order to verify that ION-F hardware would not fail at this level, the spacecraft must be designed with a 1.25 factor of safety over the 24 g level. This meant that USUSat's hardware was designed to withstand 29.75 g's of acceleration before yielding.

In addition to these factors of safety, USUSat had to make provisions for ground hardware. USUSat was designed to be the top spacecraft on the ION-F stack. This meant that some form of lifting hardware was needed on USUSat in order to successfully integrate the stack. As stated previously, USUSat was designed with four swivel rings that can be used to attach a chain lift to a crane. The rings are rated to over 600 lbs. Since, the ION-F stack is expected to weigh around 110 – 115 lbs, this provides a factor of safety over 5.0 as required for lifting hardware.

Any payload flying on the shuttle must have a minimum natural frequency over 35 Hz and preferably above 50 Hz. AFRL engineers performed a finite element analysis (FEA) and stated that to maintain the entire UN2 program at greater than 50 Hz, each stack would have to have a minimum frequency greater than 100 Hz. Since the ION-F stack resembles a cantilever beam when it is attached to the MSDS, the greatest area of concern is at the bottom of the stack, in Hokiesat (designed by VT) and the separation system. To verify the viability of this stiffness requirement, ION-F engineers performed another FEA that indicated that the ION-F stack would have a lowest frequency around 88 Hz. In response, additional stack mass was allocated to VT and UW in order to stiffen the structure near the base of the stack. AFRL also went back and added some mass to the MSDS to stiffen it. The final system frequency is unknown, but USUSat's lowest frequency is well over 100 Hz due to strength issues and is not anticipated to be a problem.

USUSat Fracture Control and Fastener Integrity

In order to make USUSat as simple as possible when fracture control was considered, all parts were designed to be non-fracture critical. All structural materials and mechanism materials were designed from Table 1 materials from MSFC-SPEC-522. These materials all show high resistance to cracking and they will all be manufactured in well characterized processes. This

ensures that any flaws will be small enough that they should not propagate significantly during the lifetime of the spacecraft.

Non-structural materials have all been designed to be contained within the structure of the spacecraft. No openings exist in the spacecraft that will allow fractured parts to escape through the structure. The glass lenses of the cameras are all contained within an aluminum barrel and cannot escape. One item of concern was USUSat's magnetic gimbal since it was possible that the device could rotate under extreme vibration. However, the kinetic energy that could be developed was extremely small and was not thought to be a concern. Finally, locking mechanisms were attached to the gimbal that would mechanically prevent rotation under launch loads, thus allowing the assembly to be classified as non-fracture critical.

All USUSat fasteners except those used by the Tini Aerospace Frangibolt actuators were obtained either through GSFC's fastener inventory program or through SDL's machine shop. The fasteners used by SDL pass through an extensive quality assurance program and have been acceptable for shuttle use in previous projects. Documentation for these fasteners will be included with the spacecraft and should not pose a problem. The Tini Aerospace fasteners are machined specifically for the actuators and meet the requirements specified for use and include paperwork from the manufacturer to allow them to be used as well.

The fracture control classifications for the major parts of USUSat are shown in Table 17.

Table 17. Fracture Control Classifications of USUSat Hardware

Component	Fracture Control Classification	Fracture Critical
Base Plate	Low- Risk	No
Top Plate	Low-Risk	No
Side Panels	Low-Risk	No
Fasteners	Low-Risk	No
Magnets	Contained	No
Stepper Motors	Contained	No
Gimbal Structure	Low-Risk	No
Camera Lenses	Contained	No
Frangibolt Actuator	Low-Risk	No
Frangibolt Bolt	Low-Risk	No
Deployable Booms	Low-Risk	No
Battery Box	Low-Risk	No
Component Boxes	Contained	No
Solar Cells	Low Release Mass	No
Patch Antennas	Low Release Mass	No

Payload Safety – Safety Data Packages and Hazard Reports

A safety data package (SDP) is responsible for documenting compliance with payload safety requirements. As a part of the SDP, hazard reports are responsible for showing the culmination of the hazard analysis process described previously. Any hazards that have not been eliminated from the system by design must be addressed in hazard reports. NSTS/ISS 13830C (NASA 1998) explains the process for payload safety reviews and the SDP submittal

requirements. JSC 26943 (NASA 1995) contains guidelines for designers to complete SDPs and hazard reports.

Hazard Reports

Hazard reports can take many forms, but the form most commonly used in NASA manned safety applications is known as JSC Form JF542B. This form includes provisions to identify the payload, hazard, affected subsystem, type of hazard, and what will be done to address the hazard. The first page of the report form is shown in Figure 49. This form will be explained in detail below.

Each hazard report generated will have similar information found in fields "a" through "i" and is shown in Figure 49. The first field is provided to give a tracking number to the hazard report. This number is generated by the payload designer and should be able to uniquely identify each separate report. This number should remain constant for the life of the payload. Field "b" is used to identify the payload or mission that the report addresses.

The phase in field "c" deals with the safety review phase that the payload is currently in. This can be Phase 0, I, II, III, or 0/I. These phases will be discussed in detail later, but in general are used to indicate whether the payload is in preliminary, detailed, or final design phases.

The subsystem field is used to identify which subsystem the hazard applies to. These subsystems are the same as those discussed previously in the chapters on space systems engineering. For NASA's manned spacecraft, the following subsystems are generally used: biomedical, caution and warning, cryogenics, electrical, environmental control, human factors, hydraulics, materials, mechanical, optical, pressure systems, propulsion, pyrotechnics, radiation and structure.

The hazard group field, "e", is used to identify what type of hazard is presented in this report. The standard hazard groups that NASA classifies hazards into are: collision, contamination, corrosion, electrical shock, explosion, fire, injury or illness, loss of orbiter entry capability, radiation or temperature extremes. Finally, the hazard should be identified as a critical or catastrophic hazard. The date field is used to identify the date the report was completed or last revised. The hazard title is used to identify the specific hazard that the report deals with.

Past these identifying fields, the next fields are used to detail the nature of the hazard and what documentation is applicable in dealing with the hazard. The hazard description should be a complete and detailed description of the hazard associated with the system. The documentation section should identify the applicable paragraph numbers from NSTS 1700.7 or from other supporting documents.

PAYLOAD HAZARD REPORT		a. NO:
b. PAYLOAD:		c. PHASE:
d. SUBSYSTEM:	e. HAZARD GROUP:	f. DATE:
g. HAZARD TITLE:		i. HAZARD CATEGORY ☐ CATASTROPHIC ☐ CRITICAL
h. APPLICABLE SAFETY REQUIREMENTS:		
j. DESCRIPTION OF HAZARD:		
k. HAZARD CAUSES:		
l. HAZARD CONTROLS:		
m. SAFETY VERIFICATION METHODS:		
n. STATUS OF VERIFICATION:		

o. APPROVAL	PAYLOAD ORGANIZATION	SSP/ISS
PHASE I		
PHASE II		
PHASE III		

JSC Form 542B (Rev November 22, 1999) (MS Word September 1997)

Figure 49. NASA hazard report form.

The hazard causes should then be listed sequentially. Hazard causes could be the environment, personnel error, design characteristics, procedural deficiencies or subsystem malfunctions (Rad et al. 1999). Each hazard report contains at least one continuation sheet that can be copied as required. Following the summary of hazard causes on the first page, each continuation sheet should receive one cause per sheet.

After the causes, the individual controls that will be used by the designers to prevent the hazard should be listed. These controls can be design features, safety or warning devices or procedures that reduce, eliminate, safe or counter the hazards arising from each cause. If procedures or processes in assembly or manufacturing are required to control the hazard, these procedures must be identified and detailed.

Next, methods for verifying the controls should be listed. These methods include tests, inspections, analyses or procedures. Finally, the status of these verifications must be listed. For preliminary reviews, most of these will be open. As the design progresses, more and more of the verification will be completed and for successfully verified controls, the status can be closed.

Any supporting documentation such as engineering logbooks or quality assurance stamped procedures can be referred to and should be available on request. The hazard reports should be included with a complete SDP. In the SDP, descriptions of each subsystem detailed in the hazard reports must be sufficient to allow PSRP members to understand the design.

USUSat Hazard Reports

From the beginning of the ION-F design, ION-F safety engineers took a proactive approach to the NASA safety process. This approach was adopted in an effort to reduce or eliminate safety related redesigns at later stages of the project. One example of this effort was in the development of hazard reports. Early in the project, GSFC engineers arranged to produce a safety workshop where students would be trained in the hazard identification and safety engineering process. ION-F engineers had taken the time to study the hazard identification process and produce draft versions of ION-F's reports. In this way, GSFC safety engineers could review the work that had been completed and adjust their teaching to cover the specific parts of training that ION-F engineers needed.

After this training session, AFRL assumed management of the UN2 program and became responsible for producing hazard reports for the entire UN2 program. While AFRL was ultimately responsible, ION-F engineers were still responsible for producing the data needed to generate the safety reports and safety data package for NASA reviews. While AFRL engineers helped in determining classification and establishing coherent formats, ION-F was still responsible for ensuring that correct information was contained in the data package and for implementing the hazard controls in the design of USUSat. The hazards that were identified in the design of USUSat are shown in Table 18. Each of these hazards had to be addressed before approval for the UN-2 payload would be approved for launch on the Space Shuttle.

Table 18. Hazards Applicable to USUSat

Hazard Title	Hazard Description	Hazard Type
Failure of Nanosat-2 Payload Structure	During launch, landing, on-orbit, or emergency landing phases, the Nanosat-2 structure fails resulting in damage to the Space Shuttle and/or adjacent payloads.	Collision
EVA Contact Hazards	During EVA, contact with sharp edges, radiation, or hot or cold surfaces could result in injury or possible loss of life.	Injury, Illness
Battery Leakage or Rupture	The release of explosive gases and/or electrolytes can lead to fire, explosion,	Fire, Explosion, Contamination,

	corrosion, contamination, potential crew injury, and damage to the Orbiter or other payloads.	Corrosion
Inadvertent Rotation of the Magnetic Gimbal / Reaction Wheel	In the event of inadvertent rotation during separation the gimbal could cause UN2 to contact and damage the SHELS thermal shroud or interfere with SHELS ejection	Collision
RF Radiation Interference with Space Shuttle Avionics, Circuitry or other Payloads	The communications subsystem inadvertently emits high-power RF radiation which may induce hazardous effects on orbiter avionics/circuitry, EMU, RMS, and/or other payloads.	RF Radiation
Inadvertent Deployment of Booms	In the event of inadvertent deployment, the booms will contact and damage the SHELS thermal shroud or interfere with SHELS ejection.	Collision

Structural Failure

Five possible causes were found that could possibly lead to failures of the ION-F structure during the mission lifetime. These causes were addressed and controls were found to prevent these failures from occurring.

First, failure could occur if the structure was unable to withstand static, dynamic and shock loads, or thermal environments. As stated previously, all ION-F structures were designed to have a minimum factor of safety of at least 1.25 on yield and 1.4 on ultimate strength under all conditions. In addition, AFRL constructed a structural verification plan that ION-F will comply with to ensure that the structures will not fail under expected loading. To verify that these controls are adequate, AFRL engineers and GSFC engineers will review and concur with ION-F structural analysis. In addition, JSC engineers will review the AFRL SVP to ensure that the plan will expose structures to expected loads.

The structure could also fail due to stress corrosion cracking. To prevent this, all structural materials must be selected from Table 1 materials from MSFC-SPEC-522B. These materials are shown to have high resistance to stress corrosion cracking. GSFC engineers will review the materials used in the manufacture of ION-F structures and concur that they conform to the standard.

The structure could also fail due to flaw growth. In response, AFRL developed a fracture control plan that would help to ensure that flaws would not develop and propagate in the structure. GSFC and JSC engineers would review the plan and results to ensure that the plan was sufficient.

Another failure cause would be due to defective manufacturing or assembly. In order to ensure that the manufacturing processes are adequate, ION-F was to develop assembly and fabrication procedures to ensure that components are fabricated correctly. In addition, ION-F had to develop certification logs that would serve to ensure that the components were assembled according to the proper procedure. ION-F would fabricate the structure according to approved

procedures using aerospace standard technologies. ION-F would use fasteners that are approved by GSFC standards and ensure that backout protection is used for all fasteners.

To ensure that these controls are adequate, AFRL and GSFC engineers would review and approve fabrication procedures and review and concur with completed assembly certification logs. ION-F would develop inspection procedures to ensure that structural parts were properly fabricated and AFRL engineers would review these inspection procedures. ION-F would use proper tracking procedures to ensure that all fasteners used are approved and will inspect to ensure that backout protection has been applied.

The final failure cause was the failure of vented containers to withstand differential pressure during ascent and descent. Any vented containers and spaces will be designed to prevent pressure build-up during ascent or descent. ION-F would conduct a venting analysis to ensure that any containers can withstand pressurization or depressurization. AFRL and GSFC engineers will review the analysis and concur to ensure that proper venting has been provided.

EVA Contact Hazards

While no EVA is scheduled to perform servicing on the UN2 payload, there is the possibility that astronauts on EVA for other missions could come into contact with the UN2 payload. While the SHELS system has a thermal shroud to help maintain proper temperatures in the system, the shroud does not fully prevent access to the stacks. Both the ION-F and 3CS stacks are taller than the shroud. As such, it is possible for astronauts to come into contact with the spacecraft.

Astronauts could possibly contact the spacecraft, bringing their Extravehicular Mobility Units (EMU) into contact with sharp edges, corners or hot and cold surfaces. To prevent this, all ION-F hardware that will be accessible was designed to prevent sharp edges using machining techniques, tape or other methods. USUSat hardware is also unpowered and therefore should not be significantly hotter or colder than surrounding equipment. To verify these design methods, AFRL and GSFC engineers will verify that ION-F hardware has been designed in accordance with approved drawings and will verify that any sharp edges or protrusions have been modified or removed. A thermal analysis will also be performed to ensure that no excessive temperatures will be present.

Astronauts could also be exposed to excessive radiation or electric shock if they were to come into contact with the spacecraft. All ION-F spacecraft use the electrical inhibit strategy discussed previously to be two fault tolerant against inadvertent activation of RF transmitters, beacon transmitters and crosslink transmitters. Pulsed plasma thrusters on Hokiesat and Dawgstar are also de-energized to preclude the possibility of electric shock. GSFC engineers will review ION-F power system design and concur that the plans provide the functionality required. ION-F must also conduct an inspection to verify that the power systems have been manufactured according to the design drawings. ION-F must also perform functional testing of the assembled power supply and inhibits and perform a final verification of inhibit status at KSC.

The last way in which astronaut contact could prove hazardous is if the structure has insufficient strength to withstand EVA induced loads. Astronauts could kick, brush or bump the spacecraft causing structural failure. To prevent this, USUSat must be designed to withstand a "kick load" of 125 lbs over a 0.5 inch diameter anywhere on the structure without failure. ION-F must perform an analysis to verify that its hardware can withstand these loads and GSFC engineers must concur with ION-F's analysis.

Battery Failure

As described previously, batteries present some of the worst hazards on the spacecraft, yet have to be present for mission success. The first major hazard that can cause battery failure is short-circuiting, both internal and external to the battery cells themselves. All cells must be initially acceptance tested to screen out cells with internal defects. The battery box and pack must be designed to electrically isolate the cells and wiring through the methods described previously. Finally, proper fusing and wiring must be used to ensure acceptable operation. GSFC and AFRL engineers must concur with the inspection and screening procedure used to eliminate defective cells. In addition these engineers must also approve the battery box design and assembly procedures. ION-F must conduct inspections to verify that the boxes have been manufactured properly and conduct mechanical and thermal testing of the box and pack to ensure that environmental conditions do not initiate shorting. Finally, AFRL and GSFC must review wiring and fusing design to ensure that adequate measures have been taken to prevent shorting.

Cell reversal and over discharging can also cause battery failure. To prevent cell reversal, the battery cells must be tested to ensure that the cell capacity is even, thus reducing the possibility of uneven discharge. AFRL and GSFC engineers must review the cell matching procedures and results. Excessive internal cell pressure can also cause failure and each cell must be individually vented to prevent pressure buildup. AFRL or GSFC engineers will review manufacturer drawings to ensure that the cells have adequate venting.

Overtemperature or freezing conditions are also responsible for many cell failures. AFRL will conduct a payload level thermal analysis to ensure that the cells will not experience temperatures above or below the manufacturer's specifications. The ION-F boxes must also be designed with sufficient control measures such as coatings, paints or heaters to maintain proper cell temperatures. To ensure that these steps are taken GSFC engineers must approve the thermal analysis and box designs; ION-F is also required to conduct inspections to ensure that the box has been manufactured according to the approved design.

The accumulation and ignition of hazardous gas mixtures, primarily hydrogen gas with oxygen must be prevented. ION-F will equip each box with two porous Teflon vent filters. ION-F must perform venting analysis to ensure that either vent is capable of relieving gas pressure under worst case conditions. GSFC engineers must review this analysis and ensure that the vents on the box have been properly located.

Finally, the batteries could leak electrolyte due to any of these conditions or other, unforeseen problems. The boxes must be filled with absorbent potting and the boxes themselves must be coated with non-conductive, electrolyte-resistant coatings. The Teflon filters must also prevent any electrolyte leakage. To ensure that these conditions are met, GSFC engineers must review box design, assembly and test procedures, and ION-F must perform inspections to ensure that the box has been properly manufactured according to these designs.

RF Radiation Interference

Premature activation of the RF system can cause serious problems by interfering with the orbiter or with adjacent payloads. As discussed previously, the ION-F electrical inhibit scheme was designed to be two fault tolerant against premature activation. GSFC engineers will review ION-F power system design and concur that the plans provide the functionality required. ION-F must also conduct an inspection to verify that the power systems have been manufactured

according to the design drawings. ION-F must also perform functional testing of the assembled power supply and inhibits, and perform a final verification of inhibit status at KSC.

Inadvertent Gimbal Rotation

Due to the design of the gimbal system, the only possible failure that could cause inadvertent rotation is premature activation of the USUSat power system. The same hazard controls and verifications discussed that prevent inadvertent RF transmission also apply to the gimbal.

Inadvertent Boom Deployment

As stated previously, electrical system failure is the cause for many hazards on USUSat. The electrical inhibit scheme described previously applies to the deployment of the USUSat booms as do the controls and verification methods. In addition, the Frangibolt actuator could deploy prematurely if the self-actuation temperature of the SMA falls within the expected thermal environment of the orbiter. The Frangibolt actuator has been designed to have an actuation temperature at least 10 °C higher than the maximum expected Shuttle environment. Tini Aerospace has agreed to perform testing to verify this before the actuators are delivered to USU.

Payload Safety Review Process

NSTS/ISS 13830C (NASA 1998) was created to define the safety review process. This process applies to both flight and ground hardware. This document describes the Payload Safety Review Panel (PSRP), the safety reviews that are required for approval of a payload, and the data that is required to be submitted for each review. The payload organization itself is responsible for assuring that its payload complies with the safety requirements detailed in NSTS 1700.7 (NASA 1989) and its sister document KHB 1700.7 (NASA 1999). These documents set the safety requirements for space flight and ground support of each payload.

Two PSRPs will be established – one for space flight and one for ground support. The space flight PSRP will be under the direction of Johnson Space Center (JSC) and the ground support PSRP will be directed by Kennedy Space Center (KSC). The panels will include representatives from program management, safety engineering, mission operations, crews, bio-medical staff, engineering directorate, and any other groups required to support operations.

The safety panel can meet for TIMs or for full payload safety review meetings. TIMs usually deal with specific issues that might not require the full PSRP to convene. The payload safety review meetings correspond with the design phases. A Phase 0 review is conducted after a systems requirements review and is designed to help identify hazards that may be present in the design to help payload designers to avoid the hazards where possible. A Phase I review is conducted after a preliminary design review (PDR). This review is intended to identify all hazards associated with the system design in this form. Often, for small payloads or payloads with relatively few hazardous functions, the Phase 0 and Phase I reviews are combined into a Phase 0/I review. The Phase II review is conducted after a critical design review (CDR) and is intended to show all the controls that have been designed to deal with the hazards identified in the Phase I review. The last review, Phase III, is designed to review the verification status of the

controls after the spacecraft has been manufactured, assembled and tested. Any controls that are unverified at this time, must be tracked separately and resolved prior to payload integration with the Shuttle.

At each review, a SDP is required. These packages require a great deal of information and the specific requirements are detailed in NSTS/ISS 13830C. In general, a detailed description of the payload and its mission are required. Any safety critical hardware must be identified and described so that the PSRP can understand its design and operation. Hazard reports must be included for systems that have been identified as safety critical. In addition, certain lists of hazards, such as all pyrotechnic devices, hazardous fluids and battery chemistries, must be included. In short, these packages should contain a sufficiently detailed description of the payload so that the PSRP can knowledgably review the design and concur or disagree with the payload designer's description of its safety.

USUSat Safety Review Process

As the integrator of the UN2 payload, AFRL was responsible for presenting an overall SDP to the safety panel at JSC. USU was responsible for providing any required inputs needed to complete the package and then to review the package to ensure that USUSat had been accurately represented. The UN2 payload has completed its Phase 0/1 review in June 2001. The program has currently changed management from the AFRL to GSFC and the Phase 2 review has not yet been scheduled.

CHAPTER 5: SUMMARY, CONCLUSIONS AND FUTURE WORK

Summary

USUSat was designed following industry standard techniques. While these techniques were modified for application toward university-designed small satellites, the same fundamental steps and processes were used.

Program requirements were drawn up which would allow the design of a spacecraft that could meet the requirements imposed upon in it by program management. These requirements were derived from launch vehicle characteristics, ejection system capabilities, launch system operational requirements and other sources. These requirements were opposed by the desire to add functionality, and a balance between increased functionality and compliance with requirements resulted in the final system.

The development of the program through various stages of design was tracked and most of the reasons for design changes were elaborated in an effort to show how overall system and safety requirements influenced the design of individual subsystems. All of USUSat's subsystems are not optimized for the tasks that they are to perform. The subsystems all had to undergo changes and compromises in order to cause USUSat to function as a whole.

As a result of some of these compromises, some requirements were not completely met. USUSat's mass is slightly over its allotted budget. In addition, the expected power consumption rates could cause USUSat to spend more time generating power and less time in formation flying missions than originally desired. However, the system as designed was able to retain functionality that will allow it to complete its primary mission objectives and to complete some secondary objectives. From this standpoint the design should be considered a success.

The design unfortunately was not completed in the time allotted and program management has shifted from AFRL to GSFC. While the project may still fly, its future is uncertain. It is certain that improvements could have been made in program management and design management phases. Some of these improvements and other lessons learned are discussed below.

Conclusions

At the end of any project, designers can often look back and make a list of items that they wish they would have known earlier or done differently in the design. USUSat and ION-F are no exceptions. While USUSat was designed to be a high-risk spacecraft, certain strategies and techniques could have reduced the risk level associated with the design.

Proper communication of program goals

The proper communication of program goals and objectives in many areas would have helped immensely. Many students who worked on USUSat were not completely exposed to the full reasons of why this project was important. Often, students thought of USUSat as another design exercise rather than as a trend setting research tool. As one example, midway through the project, a thorough program overview was given to the current group of students on the project

by Dr. Charles Swenson. Many of these students had not been present for early design phases and were working to complete designs that had been handed down from previous students. After the presentation many students remarked that the design seemed much more logical now that it had been explained properly. As a result, student morale seemed to much higher for some time afterwards and work on the project accelerated.

Also, a clear explanation of project goals among project partners would have helped. Some participants treated this project as a one-shot deal and so design work was geared solely toward completing the ION-F design. Others viewed the project as a stepping stone and tried to build infrastructure that could be used on future projects. This work often produced conflicts since schedules were lengthened when some team members tried to do more than complete a project. A clear understanding among design partners could have eliminated much of this confusion.

Establish documentation priorities and formats

As stated previously, the documentation of design status was one of the biggest difficulties in the USUSat design. These difficulties began at the start of the design process since standards for configuration management (CM) were not established early. Establishing a required amount of documentation and proper formats early in the design would have dramatically simplified the process. 3CS also had this problem and took a different approach that ION-F. ION-F decided that retaining mission functionality and completing a working design were higher priorities and so struggled through. 3CS decided to stop and establish CM procedures, and bring documentation up to a minimum acceptable level. Designers were then required to maintain this level for the duration of the project. This delay resulted in much of the functionality being lost from 3CS in an effort to meet schedules, but the design was clear and communicated effectively.

Obtain outside help in non-engineering disciplines

Related to the problem of configuration management is the need to obtain expertise in areas not usually taught in engineering classes. Bringing in students from other university departments who are skilled in areas such as program management, configuration management, and quality assurance would improve the project flow and documentation. These techniques are not taught in engineering classes, and take time and concerted effort to develop. Expecting students to develop these skills while completing technical designs in an academic environment is unrealistic. USUSat program management generally did a good job in recognizing areas where technical expertise was lacking and obtained outside help. Outside help was needed in these non-technical areas as well.

Establish technical oversight

Another change in the program that could have helped many of the students would have been to obtain the services of technical oversight personnel. Early in the project, two SDL employees, Pat Patterson and Brandon Paulson, volunteered to help students in mechanical and electrical engineering. However, these two quickly became busy with projects at SDL and since they were merely volunteering for Nanosat, it was at a very low priority. Later, Chad Fish, who

was developing the science instrumentation, began overseeing most of the students working on electrical engineering projects while the author worked with mechanical engineers. Both of us had other work and could not devote ourselves full time to helping these students. Often students could work very effectively on designs but needed a small push in the right direction to begin. Obtaining skilled people to work with the project and advise students full time would have saved much time, effort and money.

Conduct reviews with qualified professionals

Another way in which professionals could help in the design of USUSat is in helping students to understand the secondary actions that must take place to make a spacecraft successful. Professionals who are skilled in integration, test, wiring harness design, and other subjects should be brought in to help teach students how to build designs that allow these operations to take place smoothly and efficiently. AFRL sponsored a workshop where AFRL employees came in and explained these processes to students, but only two students from USU were able to attend. With the proximity of SDL to USU, it should be relatively easy to arrange for seminars to be conducted. It would also be beneficial to have students visit the work areas at SDL to gain an appreciation of the work that will be done once components have been fabricated.

Establish clear chain of authority

Another problem was in the establishment of clear lines of authority. With spacecraft design projects involving many people, any design changes or problems need to be conveyed to proper engineering authorities to ensure that proposed changes will not cause further problems later in the design process. In the case of conflicting design objectives, some authority is needed to establish the design approach that will be used. At times during the USUSat design, students would approach either one of the PI's, the mechanical systems engineer or electrical systems engineer to propose changes. Often these changes were approved but not communicated to the rest of the program management. This resulted in duplicated or neglected work and hard feelings as a result of misunderstandings. A clear chain of authority should be established in the project as well as a clear form of communications to pass the results of decisions to the rest of the team.

Establish clear work schedule for students

Students on the ION-F project were often given broad, unclearly defined objectives and tasks. Students also have the tendency to take on more work than they can possibly complete. Program management should set out a clear work schedule for students so that they know exactly what is expected of them. This work schedule should be made so that projects are realistic and can be achieved. In the event that unforeseen tasks arise that were not originally scheduled out, program management should meet with students to determine if the students are capable of completing the new tasks or if additional help will be required.

Establish clear requirements for student designs

Another problem related to documentation and work schedules is requirements flow down. Program level requirements were translated into subsystem requirements but these

requirements were often vague and not properly documented. For example, the requirements and instructions given to the gimbal designer were to "build a gimbal as small and as light as possible". If a student then designs a gimbal that has a mass of 5 kg and is as large as a basketball, how is the design wrong if that is as small and light as possible? Clear requirements must be established and documented early in the project. Designs should be evaluated on the basis of these requirements.

Establish and enforce responsibility for designs

Related to this, is the concept of responsibility for designs. Often students on the USUSat project felt that they always had more time and that the program would be extended indefinitely for designs to be completed. Senior design students felt that if they accomplished most of the work that they would pass their design class and that would be acceptable to USUSat. Along with the breakdown of work, students should be made responsible for completing designs on time. While exceptions can be made if unforeseen circumstances preclude design completion, program management should have a clear idea of the circumstances and give approval. Unfinished designs should have extensive documentation showing the problems that prevented completion and show a current status of the project so that subsequent students can finish the design as quickly as possible. Students should be treated as professional engineers and not hobbyists in this regard.

Another related concept is in the management of specific subsystems and volunteers, students and paid employees. Subsystem design leads should be paid employees or graduate students wherever possible. This can establish continuity and these individuals can be pressured to finish designs in a timely manner. When volunteers are placed in critical positions on subsystems, the possibility for mission delay increases greatly.

Establish clear, realistic schedule

When these tasks are set out, a clear, realistic schedule should be built and enforced. Schedules should take into account issues such as exams and vacations and should be tailored so that they are possible to meet. Students should be held responsible for designs but should not be expected to work 80 hours a week to meet the schedules. This schedule must also be effectively communicated to overall program management as professionals who have not been in a university environment in some time will not expect students to meet the same schedules that professionals can.

For some time during program status reviews, USUSat program management agreed with the prevailing ION-F views that designs would be performed on an Air Force established schedule. Finally, USUSat engineers sat down and created a schedule that they felt was realistic and feasible. This schedule also predicted that ION-F would not meet its delivery date by over five months. When this schedule was presented to the Air Force, they were understandably not pleased since previous communications had indicated we could meet their schedule. Clear communication of a realistic schedule early in the project would have allowed Air Force management the opportunity to plan for this reality. As it turns out, USUSat program management did not do a good job of holding students to even this schedule and it, too, has been exceeded. Realistic scheduling goals are essential.

Work safety critical designs as an early priority

This may seem to be obvious, but during the ION-F program, early schedules were reviewed to see which items seemed to be "long poles in the tent", or items that were driving the schedule length. Attention was then focused on these areas to try and complete the design in a shorter time schedule. In hindsight, attention should have been focused on subsystems that were identified as safety critical, such as deployable booms. By not initially focusing on these designs, safety concerns forced redesigns that could possibly have been avoided if more attention had been paid at an earlier date.

Maintain ability to descope project if necessary

Related to this is the need to focus on priorities and maintain the ability to descope a project if necessary. If money or time absolutely cannot be extended or if mass requirements absolutely cannot be relaxed, program management should put together a clear plan for downsizing their project if necessary. Critical projects that cannot be eliminated should then receive priority while the non-critical subsystems can be completed after critical designs have been finished.

Focus on mission success goals

While many projects start out with simple goals in mind, many experience mission growth or creep during the design. Engineers should focus on the systems that are necessary to complete main mission goals and not move to increase capability unless the subsystems that can complete mission goals are finished. While the additional functionality that can be provided by extra components and experiments can prove to be beneficial, program requirements and mission objections should not be compromised in an effort to establish additional functionality.

Future Work

While the design of USUSat was very close to completion some tasks remained to be completed. Some redesign was necessary on structural panels, the gimbal and deployable booms. Some of this work was completed after the author left the project in January 2002. Some of the issues prompting these redesigns were described in this thesis and some of the results the author has become aware of. While many of these designs are complete and most of the parts have been fabricated, the booms and some communications equipment must be finished. After this, integration and test will proceed for most of the components. Software was still being written and tested on the USUSat hardware when the author left. One large task remains in designing mission operations profiles.

Some paper work was generated but some remains to be completed for submittal to NASA Safety officials. During some of the mandated reviews detailed in the section on hazard reports, GSFC engineers noticed small problems that they were uncomfortable with. These issues are still being resolved. While the author tried to communicate effectively the inspections and paperwork agreed to during safety reviews, program management should review the requirements contained in these documents as USUSat has agreed to provide these data products as a precondition to launch on the Space Shuttle.

Overall, the design of USUSat has been challenging and gratifying at times. Students, this author included, were exposed to design situations usually not encountered in a university environment. The overall functionality of USUSat was preserved, major safety requirements were met, and most programmatic requirements were satisfied.

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